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COMPUTER SIMULATION TO ESTABLISH STRUCTURAL DESIGN AND INSPECTION CRITERIA

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16. Abstract

This paper presents a plan for FAA development of a method to establish aircraft structural inspection programs on a more rational scientific basis. The structure of a commercial air transport is inspected periodically to detect defects or damage before such deficiencies become hazardous and preferably before extensive repair is required. While the airline safety record is considered good and has been improving, there have still been accidents and incidents in recent years as a result of failures from defects which were not detected. Hazards or failure causes for this paper are classified as fatigue and corrosion; birth defects as production or design defects; and service operational or maintenance damage.

Rational development requires a model which would permit a priori evaluation of the effectiveness of an inspection program. The modeling approach selected uses computer simulation to evaluate a relatively complex model covering a major portion of the primary structure of the airplane and accounting for most of the major variables of service usage and inspection. Estimates of the benefits and cost of development show cost of development and application to a hypothetical fleet to be $316,000. Benefits were predicted as changes in the present accident rates and inspection intervals. The net benefits after deduction of increased inspection costs are estimated to be equivalent to $20,000,000 for the hypothetical fleet. Tasks, schedules, and required manpower are defined.

17. Key Words

Structural analysis; Airworthiness; Simulation; Computers; Accident prevention; Design criteria; Models; Inspection; Fatigue.

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BACKGROUND AND DEFINITION OF PROBLEM

The structure of a commercial air transport is inspected periodically to detect defects or damage before such deficiencies become hazardous and preferably before extensive repair is required. The Federal Aviation Administration (FAA) requires that the inspection be performed in accordance with an acceptable program. Initially, this inspection program is usually developed by the manufacturer and submitted by the airline to the FAA for acceptance and will specify the areas to be inspected and the frequency and method of inspection. The initial program is invariably modified as service experience is accumulated. The inspection is intensified in areas where service problems occur and is relaxed where none occur.

The airline airframe maintenance program is a massive and expensive program. Typically, the airlines will spend as much on airframe maintenance during the 20 year life of the airplane as the airplane initially cost[1]. This means that for the U. S. airline fleet of 2 to 3,000 aircraft, costing on the order of $5,000,000 each, that $500,000,000 is being spent each year on airframe maintenance, exclusive of downtime costs. It is therefore

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[1]"Jet Operating Data - B-707 and B-727," Air Transport World, September, 196[illegible], pp. 31-32.

important that this program be optimized. The FAA policy statement[1] and prevailing industry opinion would indicate that an "optimum" program would be a minimum cost program in which accident and repair costs are fully accounted for. In general, as inspection frequency and/or intensity are increased, thus increasing inspection costs, accident and repair costs are decreased, and vice versa. There are, however, as discussed below, indications or symptoms that present programs are not optimum.

While the airline safety record is considered good and has been improving, there have still been accidents and incidents in recent years as a result of failures from defects which were not detected[2]. A 707 was torn apart by a mountain wave over Mt. Fugi in Japan in 1970. The initial failure appeared to have been the failure of the vertical tail through a pre-existing fatigue crack which had reduced the original strength by one-third.

A 720 was lost in 1971 in a training accident. An engine was cut during an approach, the pilot applied hard rudder to balance the airplane, the rudder actuator broke through a pre-existing stress corrosion crack resulting in loss of control. The part in normal service was not loaded and the rudder was normally not needed except during such an engine out condition.

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[1] U. S. Department of Transportation, Federal Aviation Administration, Policy St[illegible] [illegible], Order 1000.1 (1965) pp. 7-15.

[2] Great Britain, Civil Aviation Authority, World Airline Accident Summary, (Cheltenham: Technical Publications Department), p. A/2.

A British Vanguard was lost in 1971 because undetected corrosion resulted in failure of the aft pressure bulkhead at the point where the control cables pass through resulting in fouling of control cables.

An F-27 wing failure occurred in 1968 through a pre-existing fatigue crack which appeared in x-rays of previous inspections but was not detected.

A large hole was blown out of the side of a DC-6 cabin in 1967 due to a fatigue crack which was initially detected and repaired but whose subsequent growth was not detected.

A large hole was blown out of the side of a 707 cabin in 1970 because of a large undetected fatigue crack.

A DC-3 wing failed in 1968 because of a large undetected fatigue crack.

Two British Heralds were lost in 1965 because the pressure cabin failed through an area of extensive undetected corrosion. One was being used to transport pilgrims to Mecca. They had aboard their own animal sacrifices whose wastes enroute contributed to the corrosion.

The above accidents and incidents indicate the potential benefits from reliable defect detection. Detection reliability is dependent upon: (1) the inspectability of the structure; (2) the frequency of inspection; (3) the effectiveness of the inspection method; and (4) the comprehensiveness of the inspection program. This paper will deal only with the inspection program and not the design of the airplane. Consequently, the first factor will not be considered directly.

While the above may seem to indicate the need for more inspection, it is the prevailing opinion of the operating personnel who are in charge of structural inspection that much inspection is unnecessary[1]. This opinion is apparently the result of many inspections in which no damage or only minor damage was found. Probably as a result of this view and with little other substantiating basis, inspection intervals have been steadily lengthening and the inspection sample has been decreasing over the years. For example, the overhaul interval for a mature DC-6 was approximately 6,000 hours with a 100% sample whereas a mature 727 has a 18,000 hour overhaul interval with a 20% sample.

The present inspection programs were initially developed largely by extrapolating from programs on past airplanes on an intuitive basis. While the design features were thoroughly reviewed, much available information on the structure was not used. Further, available mathematical techniques for estimating the effectiveness of a given inspection program were not used. Specifically, no attempt was made to show from a knowledge of the likelihood of a defect occurring, its rate of degradation and the frequency of inspection, that the desired safety level would be attained.

Further signs that the existing inspection programs are not optimum, especially from a safety standpoint, are: (1) from laboratory tests and service experience it is known that a defect can develop and progress to failure in less than the typical inspection period[2]; (2) the frequency of

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[1]Clarence A. Beale, Principal Federal Aviation Administration Inspector for Delta Airlines, Personal Communication.

[2]Memorandum from Chief of Engineering and Manufacturing Division to Chief of Maintenance Division on the subject of Structural Maintenance Programs, Federal Aviation Administration, Washington, D. C., 6 October 1969.

inspection is decreasing on new models while at the same time the structure is being "optimized" by eliminating areas of extra strength.

The above symptoms indicate a gap between the present and "optimum" inspection programs and indicate that perhaps the inspection program could be optimized by using all available analytical techniques to estimate its effectiveness and potential trade-offs for improvement.

A logical question is - What are the obstacles or causes for not using such an approach? The first reason is that no one knows how, at least on a practical basis. Further, many of the people involved do not have the technical training for such a task. Such technical talent is available in the FAA and the manufacturers engineering groups. The manufacturers', however, are motivated primarily only in the direction of lowering inspection costs as they must bear directly only the "design defect" portion of the accident costs. Consequently, the task of developing and promoting the broad approach falls primarily on FAA engineering, and this paper will address the problem from this latter viewpoint.

DEVELOPMENT OF METHOD FOR EVALUATING INSPECTION PROGRAM EFFECTIVENESS

Objectives and Obstacles

The effectiveness of a structural inspection program is judged by the degree to which it fulfills its purpose, which is to prevent accidents and major structural repairs. (The cost of conducting the inspection program is a separate consideration). Thus, the effectiveness of the inspection program can be quantified by estimating the probability of major structural repairs under the program. In the past there has been no attempt to make direct estimates of such probabilities. Consequently, the effectiveness of the inspection programs could be evaluated only after the fact. The rational method of estimating these probabilities would be to build a model of the system and operate the model using appropriate input data on the airplane and inspection program. The objective of this phase of the project will be to make realistic estimates of these probabilities, thus p[illegible] a [illegible] [illegible]valuation of the inspection program.

The obstacles to reaching this objective are a lack of input data and a valid model. The obstacles to obtaining such data and model are primarily:

1. A great deal of hard-to-get-data is needed to define the airplane and the inspection program adequately.
2. The "real" system is extremely complex and this makes it difficult to build a realistic model for the general case.

Possible ways of overcoming these obstacles are discussed in the following pages.

Input Data

The model of the system must be at least partially defined before the needed input data can be identified. A scenario for such a model in simple terms is:

> A detectable defect occurs thereby degrading structural strength; the defect either grows thus progressively reducing the strength, or does not grow until either: (1) it is detected and repaired; or (2) a load in excess of the degraded strength is experienced thus causing failure; or, (3) the airplane is retired from service.

Taking a lead from procedures established for reliability analysis of electronics systems[1], it would appear that the general procedure for solving this problem would be as follows:

1. Divide structure into elements.
2. Determine failure (defect) rates for each element for each hazard.

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[1] Arinc Research Corporation, *Reliability Engineering*, (Englewood, N. J.: Prentice-Hall Inc., 1967) pp. [illegible]74-2[illegible]4.

3. Determine rate of strength degradation for each element for a defect from each hazard.
4. Build a functional model of the entire structure in terms of these elements and define catastrophic failure or major repair in terms of these elements.
5. Determine rate of occurrence for each stress level for each element.
6. Define frequency of inspection for each element.
7. Determine reliability of the inspection (given that an inspection is made) in terms of degree of degradation.
8. Use these data, the functional model, and probability theory and estimate the probability of catastrophic failure and the probability of major repairs.

From the foregoing initial cut at describing the scenario and the procedures for solving the problem, it appears that the following input data for a model is needed:

1. Failure rates for each element for each hazard.
2. Rate of strength degradation for each element for each hazard.
3. Rate of occurrence for each load level of each element.
4. Frequency of inspection for each element.
5. Reliability of inspection (given that an inspection is made) in terms of degree of degradation.

Division of Structure Into Elements

Ideally, the structure or system should be divided into elements in such a way that the structure is realistically represented, the system model is simplified, and the elements are such that their characteristics such as failure rates can be easily defined. Electronic systems are quite ideal in this respect. The elements of circuits such as resistors, capacitors, transistors, etc., are easily identified and definable. They are used by the million and because of this extensive service experience on essentially identical parts under essentially the same environment, their failure rates can be rather accurately predicted. Further, failure or performance of the system can be quite accurately defined in terms of complete failure of each element, i.e., complete failure of most any tube in a TV set will disable the set.

The above situation is, to a lesser extent, also true on the black box level of electronic systems. However, the situation with aircraft structure is not so simple. The elements are not so easily identifiable nor are they repeated exactly from model to model. Further, because of extensive redundancy and because the load imposed is not constant (as is essentially the case with electronic systems) failure cannot be easily defined in terms of failure of each part nor is failure of one part independent of the failure of an adjacent part.

A review of past structural inspection programs (DC-6, DC-8, 707, 747 airplanes) indicates that typically an aircraft is divided into work areas or zones. The structural elements within each zone which are most

likely to fail and/or whose failure would significantly reduce the overall strength, are identified and singled out for special emphasis in inspection. Such elements are customarily called "structurally significant items" and the inspection program is defined for each item. The manufacturers, especially in recent years, have made extensive reviews of the design, design analysis, tests, and service experience to identify such elements. A great deal of additional effort was expended on some recent airplanes in defining the characteristics of each item in order to rate each item in terms of need for inspection. This information was presented in inspection rating sheets such as shown in Sheet 2 of Figure 1, which were developed in accordance with criteria shown in Sheet 1 of Figure 1. It is anticipated that such information will be available on major aircraft programs in the future. It is evident that manpower constraints on the project do not permit generation of new information in this area and that existing information will have to be used. Accordingly, the structure should be divided into elements which are based on "structurally significant items". It would be ideal if complete failure of any given element could be considered catastrophic as it would result in a simple chain model or series model. This could be accomplished by defining the failure mode or path for each significant structural element for each hazard and by including in the element all adjacent structure in this path whose failure would assure failure under the load equalled or exceeded each flight. Thus, the failure of any element could be considered to likely be catastrophic but some elements would consist

Figure 1

(3 Sheets)

Inspection Criticality Analysis

EXPLANATION OF STRUCTURES ANALYSIS METHOD

From 1969 Maintenance Program Proposal

	1	2	3	4
Fatigue Resistance	An indication of the fatigue resistance of the item relative to the fatigue design goal for the overall airplane.			
	Small margin above design goal	Fair margin above design goal	Considerable margin above design goal	High margin above design goal
Corrosion Resistance (Incl. Stress Corrosion)	An indication of the relative corrosion resistance of the item, considering both exposure and protection.			
	Least margin of resistance	Fair margin of resistance	Considerable margin of resistance	Highest margin of resistance
Crack Propagation Resistance	An indication of the relative ability of the material used to resist propagation of cracks.			
	Least margin of resistance (Hi Heat Treat Steel)	Fair margin of resistance (7000 series Alum)	Considerable margin of resistance (Titanium)	Highest margin of resistance (2000 series Alum)
Degree of Redundancy	An indication of the degree to which the item is backed up by redundant structure.			
	Small	----	----	High
Fatigue Test Rating	Will the loads applied to the item in the full scale fatigue test properly represent loads predicted for service usage?			
	No	----	----	Yes
Overall Rating Number (R)	A rating which considers all the above ratings and combines them by judgement into a single overall rating which represents a relative level of the structural integrity of the item.			
	1	2	3	4

STRUCTURAL INSPECTION

From 1969 Maintenance Program Proposal

SECTION ATA NO. NO.	ITEM	RATING NO. FATIGUE IMPORTANCE 1 THRU 4	CORROSION RESISTANCE 1 THRU 4	CRACK PROPAGATION (1 THRU 4)	DEGREE OF REDUNDANCY 1 OR 4	FATIGUE TEST RATING (1 OR 4)	OVERALL RATING NO. (R)		DETECTABILITY EVALUATION CATEGORY	INT CLASS NO.	EXT CLASS NO.
53-57	BS 1241 bulkhead chords, skin and stringer splices from joint with rear spar to crown centerline	1	3	2	1	4	1		IN-2-AB	1	2
57-26	Mid spar chords, web and around access openings, BLL 0 to WS 1168	2	4	2	4	4	2		IN-3-A	2	-
53-118	BS 2404 bulkhead particularly at web and chord splices, stringer S-17 to crown centerline	1	3	2	4	4	1		IN-3-A	1	-
57-25	Front spar and rear spar chords and web, general areas, WBL 127 to WS 1348; applies only to areas not duplicated in other inspection items	2	3	2	1	4	2		IN-1-AC	3	3
53-54	Skin, stringers and frames in lower lobe BS 520 to 1000 stringer S-40 to bottom centerline	4	2	3	4	4	2		IN-1-AC	3	3

STRUCTURAL INSPECTION PROGRAM

CONVERSION OF EXTERNAL AND INTERNAL CLASS NUMBERS TO INSPECTION INTERVALS

From 1969 Maintenance Program Proposal

CLASS NO.	EXTERNAL INSPECTION (INTERVAL IN FLIGHT HOURS)	INTERNAL INSPECTION (FRACTION OR PERCENTAGE TO BE INSPECTED AT 12,000 INTERVALS)
1	2,000	1/5 or 20% @ 12,000
2	4,000	1/6 or 16% @ 12,000
3	6,000	1/8 or 12% @ 12,000
4	8,000	1/12 or 8% @ 12,000
5	12,000	1/25 or 4% @ 12,000

of more than one structurally significant items and failure of one element could influence the failure of another element.

Hazards

In order to define the input data for each element for each hazard, at least the primary hazards must be defined. A brief Douglas Aircraft study on inspection programs[1] categorizes the structural hazards as fatigue, preload, and corrosion. An airline study of structural problems[2] classifies structural problems as corrosion, material, or fastener discrepancies. Jensen, Chief Engineer of Sikosky Aircraft[3], indicates that structural reliability is a function of variability in design, production, operation, and maintenance.

For the purposes of this paper, the hazards or failure causes will be classified as - fatigue and corrosion which are wear-out and aging phenomena; birth defects such as production or design defects; and service operational or maintenance damage which intuitively would seem to be random with time. There is of course, the additional hazard of complete failure due to overload but as this type of failure cannot be prevented by a structural inspection, it is not germaine to this paper.

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[1]Douglas Aircraft Co., Long Beach, Calif. "Developing the DC-10 Structural Inspection Program - Appendix." (Paper presented to DC-10 Maintenance Program Steering Committee).

[2]National Airlines Inc. "Airframe Structures Report." Proceedings of the 19[illegible] ATA [illegible] Maintenance Conference, Phoenix, [illegible]. (Washington, D. C.: Air Transport Association of America, 19[illegible]).

[3]Harry T. Jensen, "The Application of Reliability Concepts to Fatigue Loaded Helicopter Structure," Proceedings of American Helicopter Society [illegible] Annual Forum, Washington, D. C., [illegible] (New York: American Helicopter Society, 19[illegible]).

All defects detected in service are noted and to the extent recorded, their frequency and description can be determined from service records. Service records may not state the cause of the defect, although an educated guess can many times be made from the description of the defect and circumstances. If the manufacturer has issued corrective action in the form of a service bulletin, the cause of the defect may be obtained from the service bulletin as the airlines require that it include a statement of the cause . A brief description of each hazard follows.

Fatigue

This is a major hazard and failure mode which results from repeated operational loads causing a crack to initiate at a high stress point (such as caused by a hole, gauge, sharp radius, corrosion pit, etc.) and grow at a progressively faster rate until complete failure occurs. As the cumulative number of repeated loads increases with flight time, the probability of fatigue defects increases with time. The resistance of a normal structural element to fatigue cracks varies widely between apparently identical elements and the severity of the repeated loads varies widely between identical aircraft. Consequently, fatigue cracking is a probabilistic phenomenon and cannot be ruled out even when the structure is manufactured, operated, and maintained as specified. The probability of cracking under the above conditions, which will be called "classical fatigue", can be predicted from a knowledge of design characteristics and the operating environment. However, fatigue cracking under off-design conditions, such

as initiated by production, maintenance or operational errors, cannot be predicted directly from design conditions.

Corrosion

This is also a major hazard and failure mode and occurs in different forms such as surface corrosion, intergranular corrosion, and stress corrosion[1]. The susceptibility to corrosion varies for different portions of the structure depending on the corrosion resistance of the material used, the efficacy of the corrosion protection provided, and the severity of the environment. Corrosion is a progressive failure mode. In the opinion of experienced aircraft maintenance personnel[2], the probability of corrosion increases with time, in spite of the learning curve on a given model. Corrosion alone can grow to complete failure, however, it can be initiated by birth defects or service damage which can degrade the protection, lower the resistance of the material, or increase the severity of the environment. The probability of a corrosion defect cannot be directly predicted from design characteristics except where drastic measures are taken to completely eliminate a past problem. Due to normal design improvements, the probability of corrosion can be conservatively predicted from service experience on past models.

Birth Defects

"Birth" defects can be caused by production errors or design errors. Production errors such as gouges, improper radii, surface finish, corrosion protection, and heat treat are of interest. Design errors can include

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[1]British Aircraft Corporation Ltd., Viscount 800/810 Series Aircraft Manual, 15 Aug. 1967. (Weybridge, England: British Aircraft Corporation Ltd. 1967), pp. 2.12.1 - 2.12.6.

[2]Clarence A. Beale, Principal Federal Aviation Administration Inspector for Delta Airlines, Personal Communication.

errors such as under-estimating loads, neglecting to stress check a critical area, or to specify corrosion protection where protection was intended. Errors can also be made in designing repairs and production modifications. (By the definition given, classical fatigue design fatigue errors would be included under the fatigue hazard.) Birth defects are essentially one-time occurrences and do not result in progressive failure except through the action of the fatigue or corrosion phenomena which they may initiate. A survey of the type certification records of major transports shows that all have been designed under the fail safe structure design standards which require extensive structural redundancy. Consequently, with the stringent design and production quality control standards in effect, it is extremely unlikely that birth defects, which would be detected by a service inspection program, would affect enough of the redundant members to result in a complete failure in service. Such defects could result in complete service failures only through initiating one of the two basic progressive failure mechanisms of fatigue or corrosion.

Service Damage

Service damage includes operational damage such as tearing skin with a fork life; spillage of waste and other corrosive substances; and maintenance damage such as damaging parts during disassembly or assembly, tearing fuselage insulation blankets resulting in moisture collection and corrosion. It seems reasonable that such service damage is random with time. As is the case with birth defects, detectable service damage involves essentially one-time occurrences and could result in complete failure only through initiating the basic progressive failure mechanism of fatigue or corrosion. There would appear to be no way of predicting the probability of either service

damage or birth defects directly from a knowledge of design characteristics, however, past service experience may provide a basis for such a prediction.

Selection and Definition of Data Sources

There are only three possible basic sources for input data for this problem - technical analysis, laboratory tests, and service experience. As aircraft development programs are massive programs and airline experience is extensive, it would seem that these sources should be adequate. However, much of the information developed was not recorded or is not available. In some cases the only method of drawing on this background of information would be by obtaining multiple opinions of experts in the field.

As indicated by Stone[1], technical analysis for structure consists primarily of stress analysis of both damaged and undamaged structure to determine failing strength under one time loads and of fatigue analysis to determine time to initial crack or time to final failure under repeated loads. Laboratory tests to determine the same information are performed in lieu of or to validate these analysis. Repeated load tests will show the rate of strength degradation under repeated loads after a detectable crack or defect. Also corrosion tests under a severe corrosion environment will indicate the resistance of various metals and finishes to resist corrosion. The design characteristics as defined by these tests and analyses and by a design review are normally summarized in the inspection criticality rating sheets such as Figure 1.

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[1] M. B. Stone, "Developing the DC-10 Structural Inspection Program," Proceedings of the 1971 FAA International Aviation Maintenance Symposium, Oklahoma City, Okla. (Washington, D. C.: Federal Aviation Administration, 1971).

There are four basic sources of information on service defects - FAA mechanical reliability reports (MRR's), National Transportation Safety Board (NTSB) accident records, airline maintenance records, and the manufacturers records of service problems. The airline is required by law to submit MRR's to the FAA on all structural cracks, corrosion, and deformation which are beyond acceptable limits. These reports have been categorized and retained since 1962 and are readily available. They can be retrieved by computer for the last two years and manually for the previous years. Discussions with knowledgeable airline personnel indicate[1,2] that the airline retain service defect records only as long as legally required (approximately two years) and because there is no system common to all airlines, retrieval of information would be a monumental task. Consequently, airline records can be eliminated as a useful source.

NTSB structural accident records are readily available for all years of interest. As these accidents are thoroughly investigated, the cause and details on the structural descrepancies are usually known. There are only a small number of such accidents so these records would be of little use for determining structural descrepancy rates but may give a good insight to actual failure modes and circumstances for catastrophic failures.

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[1] Edward L. Thomas, Director of Engineering, Air Transport Association of America, Personal Communication.

[2] H. L. Gullickson, Airframe and Systems Engineering, American Airlines. Personal Communication

It is airline practice[1] to report failures and to send failed parts to manufacturers so that they can determine causes and fixes. As there are only two major U. S. manufacturers for transports of recent vintage, these records are relatively concentrated. Also, in the last two years (e.g., Boeing 747) a real effort has been made to assemble such information into a readily accessible data bank. The manufacturers records are likely to provide a better definition of failure causes than the MRR's because of the failure investigation but have the disadvantage that they are generally not automated and are not readily available to the FAA because of their proprietary nature and the cost the manufacturer would incur in making them available. However, manufacturers do issue service bulletins outlining "fixes" for significant structural problems and at the airlines request, these bulletins identify the cause of the problem. These service bulletins are submitted to the FAA for approval and are therefore available. It would appear that the manufacturers records directly or through service bulletins would be a good source for identifying the cause of the defects for which the MRR's are not adequately definitive. At least one manufacturer has indicated that he would provide information if formally requested and the workload was not excessive[2].

The above discussion provides a general basis for the following detailed definition and selection of data sources for input data for the individual system variables.

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[1] Henry J. Young, "Aircraft Structure Reliability Programs," Proceedings of FAA Maintenance Symposium (Washington, D. C.: Federal Aviation Administration, 1966), p. 5.

[2] Richard Meinert, Assistant Airworthiness Manager 707/727/737, The Boeing Co., Personal Communication.

Defect Rate

As previously discussed under hazards, it cannot necessarily be assumed that the rate of occurrence of a given type of defect remains constant throughout an airplane's service life. Some hazards, such as fatigue and corrosion, increase with time in service while other hazards, such as service damage and birth defects, tend to decrease with time in service or remain constant. A survey of the MRR's on a given jet model for the period of 1962 to 1972 is summarized in Figure 2 and confirms the above trends. The data sources for the defect rates and their variation with time for the various hazards are discussed below.

Classical Fatigue Defect Rate

During the design and certification of each major new model, the mean fatigue life of each significant structural element is determined by fatigue test and/or analysis for the expected spectrum of repeated loads.[1] On recent models, this information has been available to the FAA in the documentation justifying the proposed inspection program. With this information on the design and the information from the literature on the variation of fatigue life and loading spectra between individual airplanes, it is possible by the method described by Anderjaska[2] to estimate the probability of fatigue cracking as a function of the number of flights or flight hours completed. Use of this method would result in a rate that increases with the flight time on the aircraft.

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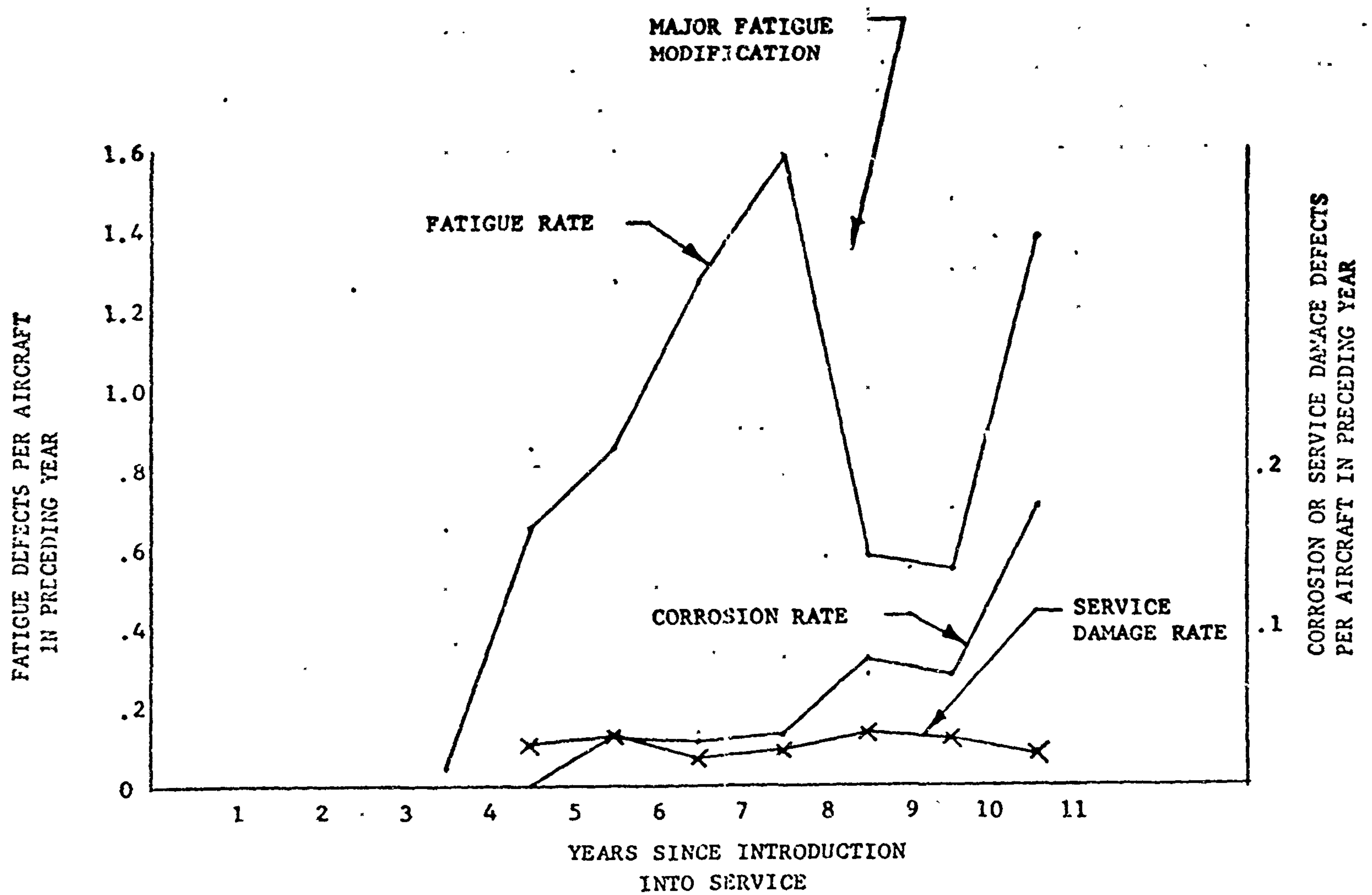
[1]Stone, "Developing the DC-10 Inspection Program," pp. 4-5.

[2]Arnold E. Anderjaska, Fatigue Substantiation Procedures for General Aviation Aircraft Structures, SAE Report [illegible] (New York: Society of Automotive Engineers, 1971).

Figure 2

Typical Defect Rate Trends



MAJOR FATIGUE
MODIFICATION
FATIGUE RATE
CORROSION RATE
SERVICE
DAMAGE RATE
FATIGUE DEFECTS PER AIRCRAFT
IN PRECEDING YEAR
CORROSION OR SERVICE DAMAGE DEFECTS
PER AIRCRAFT IN PRECEDING YEAR
YEARS SINCE INTRODUCTION
INTO SERVICE
0
.2
.4
.6
.8
1.0
1.2
1.4
1.6
.1
.2
1
2
3
4
5
6
7
8
9
10
11

If the probability estimates are based on a fatigue life determined by fatigue analysis in lieu of test, the possible errors of the fatigue analysis would have to be accounted for as fatigue analysis is notoriously inaccurate. This lack of reliability was documented in an extensive study of the subject by the Royal Aeronautical Establishment[1]. The probability of the fatigue analysis under-estimating or over-estimating the true mean life by a given amount could be estimated by the results given in the above study and a correction made on a probabilistic basis.

The classical fatigue defect rate could be estimated from service experience but except for early problem areas, the rate could not be established until late in the life of the fleet. With this approach, the inspection for a given problem area could not be established until after the fact unless based on the experience of previous models. Fatigue tests[2] show that minor design changes can result in large changes in mean fatigue life and the probability of a fatigue defect varies drastically with mean fatigue life[3]. As most new models have an improved fatigue design, use of service experience on past models have an improved fatigue design, use of service experience on past models to predict the capability of new models would be unrealistic. For these reasons, it is concluded that prediction of the classical fatigue defect rate directly from design characteristics would be more effective.

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[1]K. D. Raithby, A Comparison of Predicted and Achieved Fatigue Lives of Aircraft Structures, Technical Note No. Structures 301, (Farnborough, England: Royal Aircraft Establishment, 1961).

[2]Stone, "Developing the DC-10 Inspection Program," Fig. 3.

[3]Anderjaska, Fatigue Substantiation Procedures.

Non-Classical Fatigue Defect Occurrence Rate

This category of defect involves fatigue cracks which initiate as a result of other hazards such as birth defects, service damage and corrosion. Rates for this type of defect can be established for a given type of element on past models for the period subsequent to 1962 by reviewing the MRR's to determine the number and nature of defects and by referring to the manufacturers service bulletins and records, as necessary, to identify the cause of the defect. These rates would apply to similar new models except in cases where a design change has been made to a new model to eliminate a particular type of initiating hazard. In these cases, the rate could be reduced by eliminating appropriate service defect reports from the rate calculation. As there have been no analytical techniques developed to predict this category of defect directly from design characteristics, use of service experience is the only available alternative. The design characteristics as defined in the manufacturers inspection criticality rating analysis can be used to categorize elements according to their susceptibility to the hazard of corrosion.

Corrosion, Birth and Service Damage Defect Occurrence Rate

Under the approach described above, only corrosion, birth, and service damage defects which had initiated fatigue cracks at the time of detection in service would be considered in calculating the non-classical fatigue defect occurrence rate. Consequently, such defects which were reported in service but had not yet initiated fatigue cracks at the time of detection (e.g., loose fasteners) must be accounted for separately. Such defects could eventually cause fatigue cracks and thus effectively lower the mean fatigue life.

The effect of the most common and significant types of these defects (e.g., fretting resulting from loose fasteners, corrosion pitting, rough surface finish, etc.) on mean fatigue life can be obtained from fatigue research reports[1,2,3,4,5,6]. If the defects occur in a high stress area, such as a rivet hole, the effect on mean fatigue life would correspondingly be greater. This type of defect could be accounted for by estimating the rate for each element from service experience and by estimating its effect on the mean fatigue life of the element.

Corrosion Defect Occurrence Rate

Corrosion is a progressive failure mode which can result in complete failure without initiating a fatigue crack, especially if the corrosion environment is severe. Thus corrosion involves an additional hazard to the hazard of initiating a fatigue crack. The rate for occurrence of defects due to this additional hazard can be estimated from service experience as in the previous paragraph but should be categorized in terms of severity of the corrosion environment.

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[1] Naval Air Engineering Center, Aeronautical Materials Laboratory, Corrosion and Fatigue Evaluation of Spar Cap Specimens from HU-16 Wings, Report No. NAEC-AML-[illegible] (Philadelphia: Naval Air Engineering Center, 1967).

[2] M. S. Rosenfeld, Effects of Corrosion on Fatigue Life (Johnsville, Penn.: Naval Air Development Center, 1969).

[3] Herbert H. Leybold, Herbert F. Hardrath, Robert L. Moore - An Investigation of the Effects of Atmospheric Corrosion on the Fatigue Life of Aluminum Alloys, [illegible] (Washington, D. C.: National Advisory Committee for Aeronautics, 1958).

[4] Aeronautical Systems Division, Effect of Corrosion on the Fatigue Behavior of 2024-T4 Aluminum Alloy, ASD Technical Report 61-121 (Dayton, Ohio: Aeronautical Systems Division, 1961).

[5] Battelle Memorial Institute, Environmental Corrosion-Fatigue Behavior of Aluminum Alloys, DMIC Memorandum 2[illegible] (Columbus, Ohio: Battelle Memorial Institude, 1970).

[6] Sikorsky Aircraft, Fatigue Properties and Analysis, Report No. SER-5058[illegible] (Stratford, Conn.: Sikorsky Aircraft, [illegible])

Rate of Strength Degradation

The occurrence of a detectable defect causes little reduction of strength in a typical transport structure because of its extensive redundancy and tolerance for damage. Such a defect must grow in size to cause a significant reduction in strength. As previously mentioned there are two basic mechanisms of growth, fatigue or corrosion or a combination of the two. The data sources for rates of strength degradation under these mechanisms are discussed below.

Rate of Strength Degradation Due to Fatigue

As previously mentioned, for recent major transports, the manufacturers have rated the inspection criticality of each structurally significant item in terms of major variables. As shown by Hardrath[1], two of these variables, material crack propagation resistance and the degree of redundancy in construction are the major variables in determining the rate of degradation due to fatigue. Consequently, these ratings provide a good method of categorizing the various structural significant items. The rate of degradation can then be determined for each category from the usual development tests of structure under repeated loads such as shown in Figure 3 for the DC-10[2]. It is understood that the degree of redundancy rating has been based on the safety margins shown in the proof of strength with obvious damage that is required by FAA regulations[3].

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[1] Herbert F. Hardrath, Fatigue-Crack Propagation and Residual Static Strength of Built-Up Structures, Technical Note 4012 (Washington, D. C.: National Advisory Committee for Aeronautics, 1957).

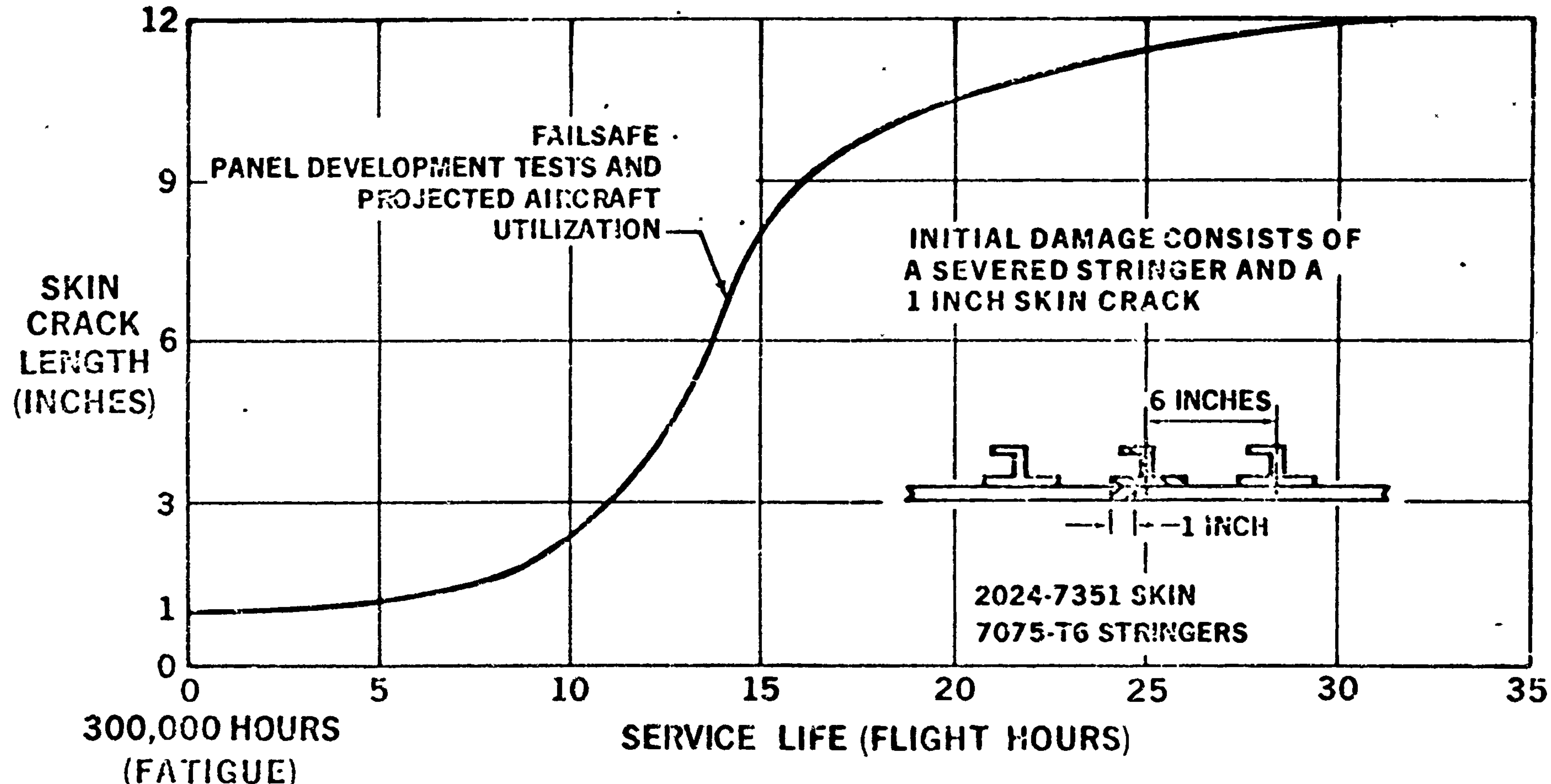
[2] Stone, "Developing the DC-10 Structural Inspection Program," Fig. 7

[3] U. S. Department of Transportation, Federal Aviation Administration, Code of Federal Regulations, Title 14, Part 25, Jan., 1970. (Washington, D. C.: Government Printing Office, 1970), para. 25.571.

Figure 3

Typical Crack Propagation Test Result

LOWER SURFACE SKIN CRACK PROPAGATION



PR71-GEN-21601

Service reports can also provide information on the rate of degradation. Service reports many times give the size of defect detected and the time since the last, presumably negative, inspection. From this information, the rate of growth can be deduced. Also, in some cases a detailed metallurgical exam of the failure[1] is made under high magnificantion and the crack growth rate can be conclusively established. While this service information is directly applicable to the structure in service and is of limited use on new supposedly improved models, well documented cases could be used as a check on the rate of degradation as determined from laboratory tests on similar structure on new models.

Rate of Strength Degradation Due to Corrosion

In the inspection criticality rating, each structurally significant item is rated in terms of resistance to corrosion taking into consideration both exposure, protection and basic resistance of the material. These factors are the major variables both for time to initiate and time for growth of corrosion defects. The resistance of the structure or the severity of the corrosion environment can be affected by birth defects or service damage which are not anticipated in the rating. The severity of the situation is indicated by the time required to initiate corrosion.

As indicated under rate of occurrence of corrosion defects, little realistic information on corrosion performance is available from laboratory tests and therefore service experience is the primary data source. As explained under rate of degradation for fatigue, the rate of degradation for

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[1]Royal Aircraft Establishment, A Metallurgical Examination of Pieces of Freight Door [illegible] from [illegible] (Incident - 24 Sept., [illegible]) (Farnborough, England, Royal Aircraft Establishment, 1970).

corrosion could be estimated from service reports of corrosion for structure similar to each "structurally significant item". It would appear that the rate of degradation could be expressed as a function of corrosion resistance rating, which is primarily a function of material, location and time to initiate corrosion. Well documented cases of service corrosion could be used to define the functional relationship.

Rate of Load Occurrence

In order to estimate the probability of a degraded structure failing due to the occurrence of a load in excess of its strength, the rate of occurrence must be known for loads of various magnitude for each element. For airplane structure, these loadings primarily result from gusts, maneuvers, and cabin pressurizations. Fortunately much work has been done in this area. NASA has, for many years, conducted a program on transports of measuring accelerations at the airplane center of gravity (C.G.) caused by gusts and maneuvers. The speed, weight, and altitude associated with each acceleration occurrence are also measured. The rate of equalling or exceeding each magnitude of C.G.acceleration can be estimated for new transports from these data on past transport such as illustrated by the curves of Figure 4 which are conventionally called a "loading spectra". However, the load on an individual element (except pressurization loads) varies not only with C.G. acceleration but also with airplane speed, altitude, weight, and C.G. position. Derivation of the element loads from this basic data is very involved and beyond the scope of the project. However, such element loads are normally derived by the manufacturer during his fatigue analysis program. The detailed analysis is not directly available to the FAA as a

fatigue analysis is not required by the FAA for fail safe structures and the manufacturers have been reluctant in the past to furnish the FAA with copies of these analyses. Furthermore, as Larsen[1] indicates, the fatigue analysis is usually computerized with the derivation of the element loading spectrum an intermediate step to obtaining the final output which is the mean fatigue life of the element. As a result the loading spectrum for each element is not directly available from the computerized fatigue analysis. In view of the resource constraints on this program it would not be feasible for the FAA to obtain element loading spectra from this analysis.

Lockheed and Boeing, under a $250,000 FAA contract, have calculated the gust loading spectra for major components, such as the wing, fuselage, and tail, for several typical jet and piston aircraft[2]. These spectra were expressed in terms of a ratio to design limit load and are illustrated in Figure 5 (FAA regulations require that the structure be capable of supporting design limit load without detrimental deformation and 150 percent of limit load without failure). Design limit load as shown in this work is roughly the maximum load an average airplane will experience in its lifetime. As limit load on an element is defined as the most critical combination of C.G. acceleration, speed, altitude, and center-of-gravity position within the design envelope[3], it will occur much less frequently than indicated for the design maximum center-of-gravity acceleration in Figure 4.

---

[1] A. C. Larsen and R. E. Watson, "State of the Art in Design and Testing to Ensure Continued Aircraft Structural Integrity," Proceedings of [illegible] Maintenance Symposium (Washington, D. C.: Federal Aviation [illegible], [illegible]).

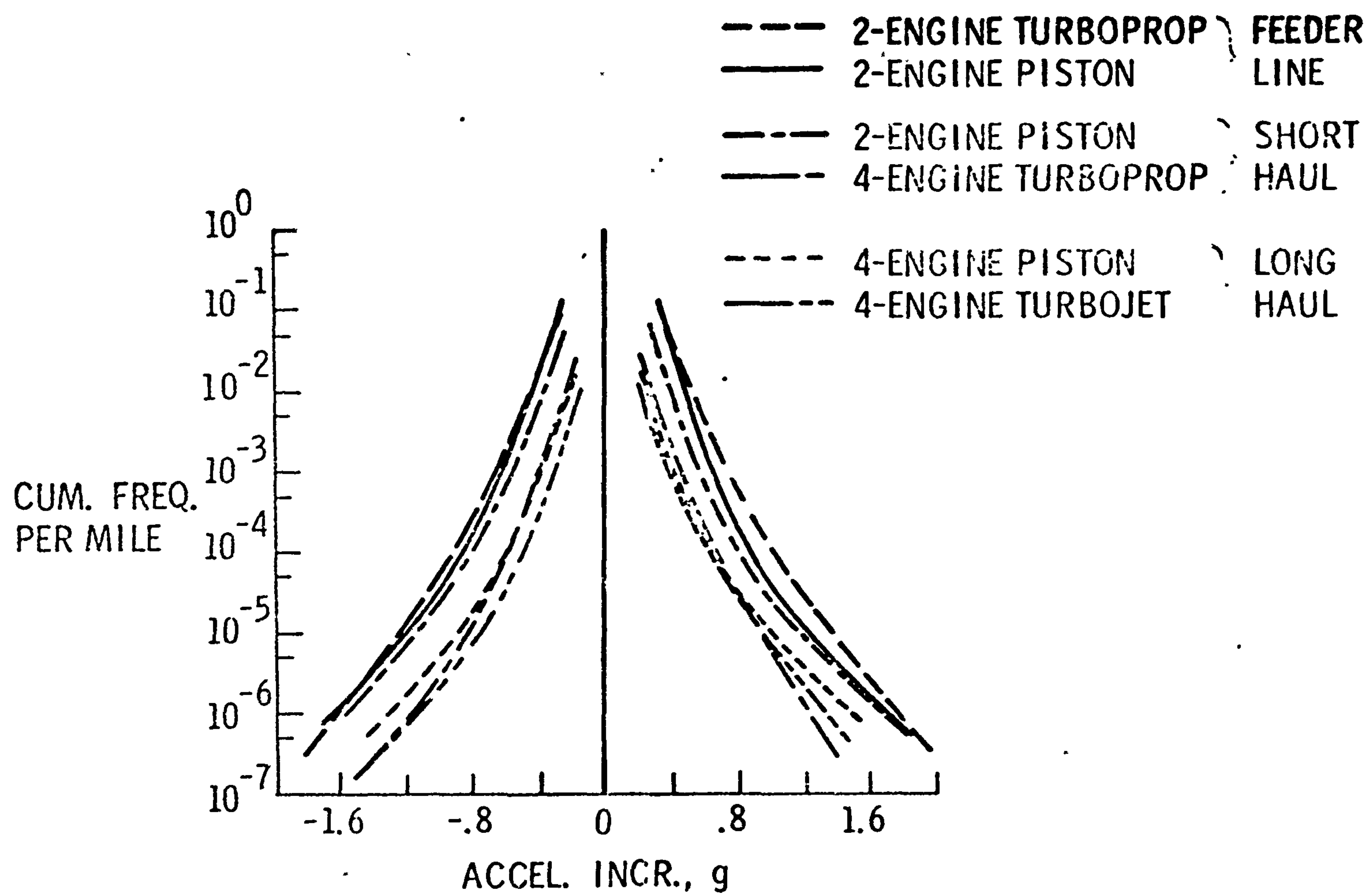
[2] Federal Aviation Administration, Development of a Power-Spectral Gust Design Procedure for Civil Aircraft, [illegible] (Washington, D. C.: Federal Aviation Administration, [illegible])

[3] U. S. Department of Transportation, Code of Federal Regulations, para. 25.301.

Figure 4

Typical Airplane Loading Spectra

(Reference-Trends in Repeated Loads
On Transport Airplanes, NASA, Langley)



NASA

Total in-flight acceleration distributions.

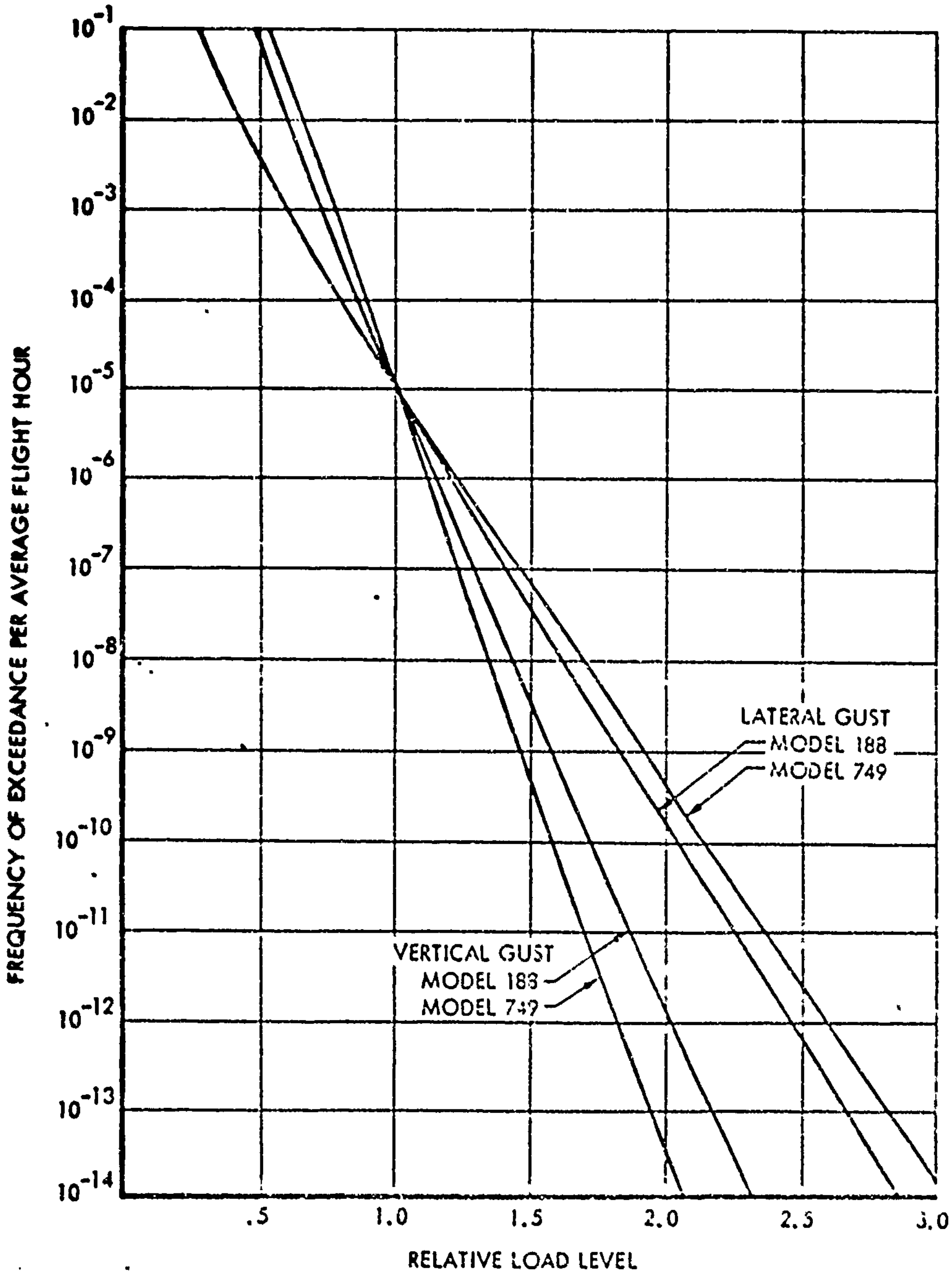
If it is assumed that the loading spectra derived in the above study are reasonable for the major components of new models, the loading spectra for individual elements of the major components can be estimated by adjusting the loading spectra to account for any excess strength present in each element. For example, if an element was capable of supporting 300 percent of design limit load without failure in lieu of the 150 percent required, then design limit load would occur at the same frequency as indicated for the 150/300 or 50 percent of design limit load in the above study. As it is difficult to design a structure with exactly the required strength, there is usually some excess strength available.

The above assumption is not unreasonable because current new models (except for the supersonic transport) are flying at the same speeds and altitudes, are of similar construction and used similar design practices as past models. Further NASA data shows that for typical jet transports, the loads resulting from maneuvers are insignificant compared to those resulting from gusts. So the above study is considered a reasonable first source of input data for rate of load occurrence for gusts and maneuvers. Use of loading spectra from the manufacturers' fatigue analysis may be feasible at a later date should the manufacturers indicate they would provide the loading spectra at no charge or should additional money become available for contracting the manufacturers to provide this data. Pressurization loads occur once per flight and are defined in terms of cruising altitude. The frequency of occurrence of the various cruising altitudes can be easily obtained from NASA data on transports.

Figure 5

Typical Variation of Load Level
With Frequency of Exceedance

(Reference-Development of a Power-Spectral Gust
Design Procedure for Civil Aircraft, FAA, Washington)



TYPICAL VARIATION OF LOAD LEVEL
WITH FREQUENCY OF EXCEEDANCE

Inspection Reliability

The probability of detecting a defect during a scheduled inspection depends upon how well the inspection is performed and on how obvious the defect is during the inspection. The latter variable depends on the size and type of defect and on other factors such as whether the defect causes a fuel leak or whether adjacent structure holds the crack closed.

There are no analytical techniques for directly predicting the probability of detection of a given defect during an inspection and a review of the literature does not reveal any significant laboratory studies on the subject. During some laboratory fatigue tests[1] it has been noted that cracks of a certain size are discernible by certain techniques. Further, during most such tests, each crack detected is reported along with its size, location, and the method of inspection. However, this same information along with the time since last inspection is given in Mechanical Reliability Reports on service aircraft. As service data is extensive and is much more realistic than that from laboratory inspection, it is the best source of data. The problem with such data is that while it indicates the size and type of defect that can be detected, it does not report cracks that were missed in inspection. Such information could be deduced from a knowledge of crack growth rates and the time since the last inspection but it would be a monumental task beyond the resources available for this project. Further, as little crack growth data has been available for past models, it would be

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[1]J. Schijve, et al. Fatigue Tests with Random and Programmed Load Sequences with and without Ground-to-Air Cycles. [illegible] on Full-Scale [illegible]. (Amsterdam, Holland: National Aerospace Laboratory, [illegible]), Appendix C.

highly inaccurate. It would be possible to determine the size of the missed defect for a few well documented cases where crack growth data was readily available either from laboratory tests or an examination of the fracture. For example, in one case of an in-flight decompression, the fracture surface was examined under high magnification[1] and the number of loading cycles (in this case pressurization cycles, once per flight) were counted back to the last inspection thus accurately establishing the size of the crack missed in the last inspection. The probability of detection could not be established from these few cases but they could be used as a guide.

Estimating the probability of detection of a defect during inspection is a very subjective matter which is difficult to judge directly from a few completely defined cases and many relatively undefined cases without actual inspection experience. As there are many inspectors who have had many years of experience in precisely the area concerned, it would seem appropriate to make a survey such experts to obtain a multiple opinion estimate of the probability of detection for various size defects under various circumstances and inspection methods. The experience of such experts encompasses much unrecorded and intangible information that would not be available from any other source. The few well-documented cases of undetected as well as a summary of cases of detected defects could be furnished to them to assist in their estimates.

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[1]Royal Aircraft Establishment, A Metallurgical Examination of Freight Door Skinning and Frames.

The manufacturer's inspection criticality rating analysis can also provide some additional information on detectability. In this analysis, each structurally significant item is rated on whether a defect could cause a fuel leak, whether the item was ten feet or more from the ground, and whether internal damage could be detected by an external inspection. These ratings (as is apparent from the discussion under inspection frequency) are in effect defining the levels of inspection in which a defect could but not necessarily would be detected. This information can be used in development of cases to be posed in the multiple opinion survey.

Visual inspection is the primary inspection method used by the airlines and the primary type considered in this project. However, it would be desirable if the model and multiple opinion survey would allow for consideration of supplementing visual inspection by other inspection methods such as x-ray, eddy current, and use of penetrant aids. The actual inspection with these alternate methods is more costly than a visual inspection but the cost of gaining access (such as disassembly of parts) may be less expensive and/or the probability of detecting smaller defects may be increased.

Inspection Frequency

Any of an infinite number of inspection schemes and frequencies could be chosen as input data for the model but from a practical standpoint, the evaluation has to be limited to typical inspection programs and alternates thought to be feasible by experts in the field.

A "typical" inspection program as synthesized from descriptions in various papers and documents[1,2,3] and discussions with knowledgeable people is a multi-level program as follows:

A cursory walk-around inspection every 8 to 25 flight hours. These inspections are typically entitled - enroute service, turnaround inspection, or terminating pre-flight check. Discrepancies such as obvious fuel leaks could be detected in such inspections. Pre-flight inspections are also made but with modern docking systems and night flights, these are relatively ineffective.

A close walk-around inspection every 50 to 100 hours. These inspections are typically entitled - service checks, A or B checks or area checks and involve 6 - 12 manhours, much of which is spent on servicing systems. External defects in structure less than 10 feet from the ground as well as significant fuel leaks could be detected in this type of inspection.

A close walk-around inspection plus a close inspection of certain external and easy access areas every 500 to 1,000 hours. These latter areas are selected because they are critical and easily

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[1] C. A. Ansvitt, H. F. Heap, and H. L. Storey, "Aircraft Structural Sampling Inspection Programs." Proceedings of FAA Maintenance Symposium. (Washington, D. C.: Federal Aviation Administration, [illegible]).

[2] Federal Aviation Administration, Maintenance Review Board Report, Boeing 747 (Washington, D. C.: Federal Aviation Administration, [illegible]).

[3] Federal Aviation Administration, Maintenance Review Board Report, McDonnell Douglas DC-10-10 (Washington, D. C.: Federal Aviation Administation.

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Inspection Frequency

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accessible and usually involve control surfaces and hinges, wheel wells, and fixed structure surrounding major doors. These inspections are typically entitled maintenance checks, C checks or area checks and require 400 to 500 manhours most of which are spent on powered or mechanical systems and not on fixed structure. From the standpoint of fixed primary structure, this level of inspection differs from the previous lower level in that access is gained by means of stands, etc., to structure surrounding control hinges and major doors which are more than ten feet from the ground. Consequently, defects could be detected in this additional structure in this inspection.

A close inspection of the complete exterior of the airplane every 2,000 - 8,000 hours. Access to these areas is gained by use of work stands, "cherry pickers", etc. Any external defect could be detected in this inspection.

A close sampling inspection of a certain percent of aircraft every 7,000 to 16,000 hours of internal structure to which access is gained through access doors, draining of fuel, removing cabin liners, etc. At certain points, sealant, fasteners, etc., may be removed and inspection aids used. Structural defects in internal areas could be detected in this level of inspection.

Development of Model

E. S. Quade[1], defines a model as "a simplified, stylized representation of the real world that abstracts the cause-and-effect relationships

[1]E. S. Quade, System Analysis Techniques for Planning-Programming-Budgeting. (Santa Monica, Calif.: Rand Corporation, 1966).

essential to the question studied." He indicates that the means of representation may range from a set of mathematical equations or a computer program to a purely verbal description of the situation in which intuition alone is used to predict the consequences of various choices.

It is apparent that an infinite number of different models could be developed for any situation. The dilemma faced in developing a model is that to make the model realistic in all respects, it tends to be complex and yet the more complex it is, the more difficult it is to evaluate. A common method used to simplify the model is to consider only the primary relationships affecting the problem being analyzed. The objective is to develop a model which is as realistic as necessary to solve the problem and yet can be evaluated within the constraint of the resources available. The general problem of estimating the probability of aircraft failure, which is the purpose of this model, has been studied from two diverse approaches. Eggwertz[1], in his study of inspection intervals for fail-safe structures, predicted the probability of failure of a typical small section of a wing under various simple inspection schemes by means of a rigorous manual mathematical analysis supplemented by a computer analysis of some of the more complex aspects using Monte Carlo and Step Integration procedures. From this study, certain conclusions were drawn for the entire structure under the operational situation.

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[1]Sigge Eggwertz and Goran Lindsje, Study of Inspection Intervals for Fail-Safe Structures (Stokholm, Sweden: Aeronautical Research Institute of Sweden, 1970).

A quite different approach was planned for the fault tree analysis of the Boeing Supersonic Transport[1]. It was planned that the entire airplane would be analyzed for all accident causes and the probability of catastrophic accident be estimated using the fault tree analysis technique evaluated by computer simulation. A fault tree is essentially a logic diagram. Simulation was planned because the analytical approach was considered too time-consuming, laborious, and error prone for evaluating fault trees of the complexity envisioned. While admittedly a much more ambitious problem was being solved, the magnitude of effort planned was much greater than expended by Eggwertz in his study.

In one case, a small but typical part of the problem was evaluated by a rigorous mathematical analysis of a simplistic model and the results of this evaluation was extrapolated to draw conclusions about the whole problem. In the other case, it was planned that the whole problem be analyzed by use of realistic but large and involved model which could be evaluated only by computer simulation. In the first case, conceptually the model was simple but the mathematical solution would not be easily understood by someone not well versed in mathematics. However, in the second case, the basic element of the model is a logic diagram which along with the simulation method of solution can be easily understood by a layman.

For the project being planned in this paper to be effective, the conclusions must be realistic and valid and accepted as such by the

[1]Boeing Co., SST Fault Tree Analysis, Document No. D6A10784-1 (Seattle, Wash.: Boeing Co., 1969).

operational personnel who establish and carry out the inspection program. The situation being modeled is an exceedingly complex one involving many relationships essential to the question being studied. The conclusions will not be accepted if the model is not understood and/or if it is apparent that there are important factors affecting the effectiveness of the inspection program which are not considered in the model. The large apparent discrepancies between actual practice and inspection programs indicated by simple models have been discussed extensively by the author with operational personnel. Their reactions are:

1. The model was not realistic as it does not account for important factors such as corrective action and supplemental inspections; or,
2. The complex, flexible, multi-level actual inspection programs may be equivalent to the simple periodic inspection indicated to be necessary by the model; or,
3. The model must be wrong because service experience is the real proof and based on their limited view of it, service experience has been good.

It is apparent from the foregoing, that the model used should be realistic and easily understood. It appears that approach planned for the supersonic transport fault tree analysis of using a logic diagram evaluated by computer simulation would be most effective for the problem at hand. The actual model could be restricted in size to stay within resource constraints by analyzing only part of the structure.

Definition of Structural Elements

This topic has been discussed previously in a general way under a similar heading. However, it now is desirable to be more specific. It would be ideal if each structurally significant item defined in the criticality analysis (Figure 1) were homogenous, independent, and not redundant (i.e., its failure would be catastrophic). If this were the case, each structurally significant item could be considered an element. Unfortunately, this is not the case with an airplane structure.

This study is concerned with only the more important or primary structure of the airplane whose failure is likely to be catastrophic. On a typical airplane this structure consists of the fuselage shell (skin, stringers, and frames), the box beams in the wing and empennage surfaces, and the attachments of empennage and wing to the fuselage. The fuselage shell acts as a beam carrying the flight loads imposed by the cabin payload, as supported and guided by the wings and empennage. The wing and empennage box beams carry the flight loads to the fuselage as beams to support and guide the fuselage through the fuselage attachments. Experience has shown that the loss of capability of these structures to carry flight loads is normally catastrophic. The fuselage shell also acts as the cabin pressure vessel. While loss of pressure containment capability is not usually catastrophic, massive failures resulting from explosive decompression may be catastrophic because of loss of vital systems in the fuselage shell or because the flight load carrying capability of the fuselage has been destroyed.

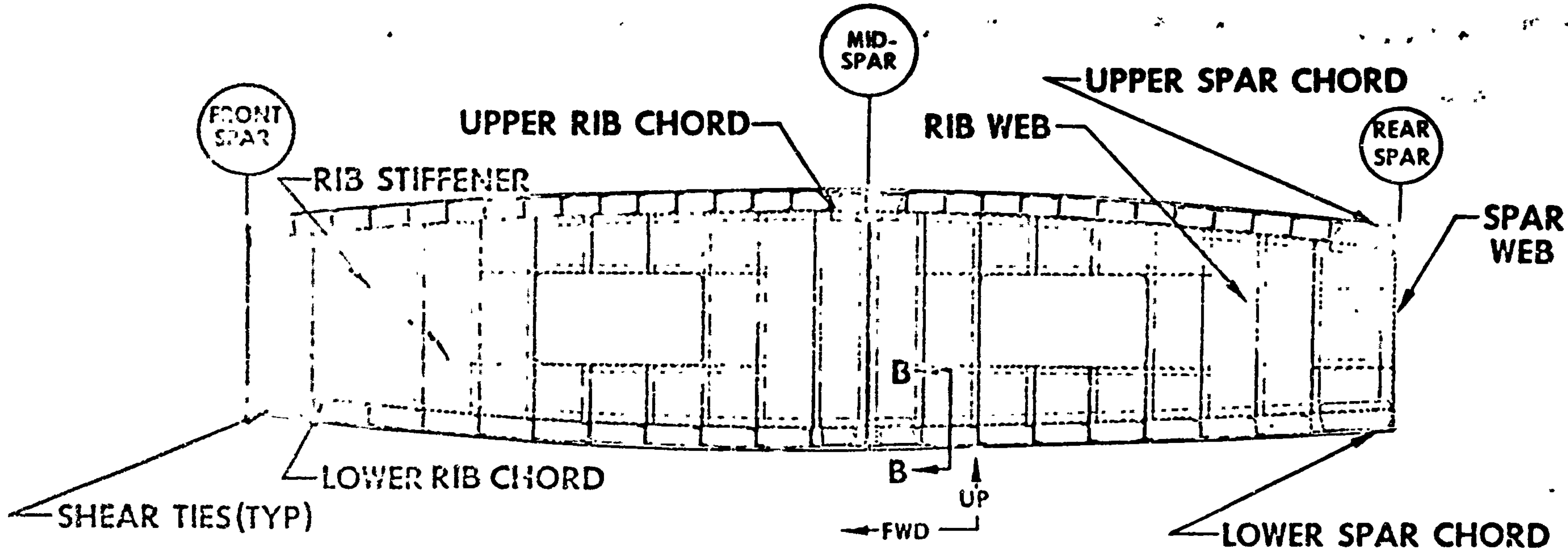
The flight loads produce internal bending, shear, and torsion on the wing, empennage, and fuselage beams. A cross section of a typical wing box beam is shown in Figure 6. Failure of such a chordwise section would be catastrophic. Typically, the bending loads are carried by the top and bottom covers (skin, stringers, and spar caps), the shear by the vertical webs of the spars, and the torsion by the boxes formed by the skins and spar webs. As bending and torsion are carried by essentially the same structure and a bending failure will typically occur with less structure failed, the bending mode of failure could be considered to control. Service and test failures show that wing bending is by far the most common failure mode. A pure shear failure under normal flight loads would probably involve all three of the vertical webs which are physically separated. As defects which grow fore and aft (i.e., chordwise which is normal to the principal stress) are the primary concern here, a defect in one spar web would normally have to grow through an adjacent cover to reach an adjacent spar web. Consequently, a pure shear failure is not too likely.

In view of the above situation, to obtain an element which is not redundant and is independent, the element would have to include a chordwise slice of the complete wing box as shown in Figure 6. This would include typical failure paths and is necessary for independence because failure in one part of the cross section affects adjacent members. Fortunately, except as discussed later, one chordwise slice of structure is relatively independent of defects in adjacent inboard and outboard slices, and encompass the most common wing failure path. However, to ensure that the strength degradation of multiple defects in the element are additive, the chordwise slice

Figure 6

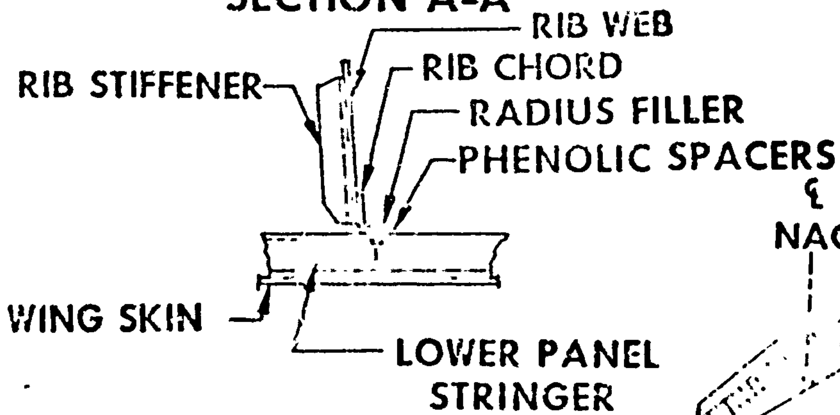
Cross Section of Typical Wing Box Beam

(Reference - Boeing 747 Structures Brochure)

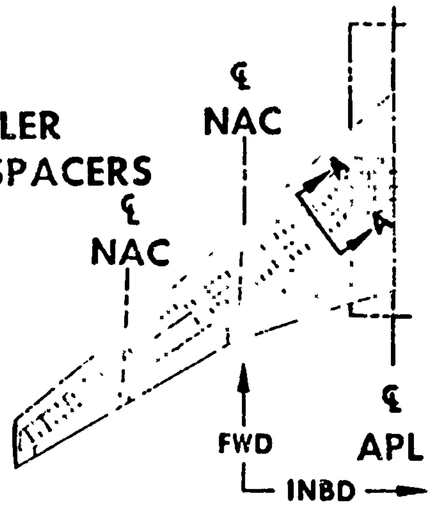


SECTION A-A

WING TORQUE BOX	
ALUMINUM MATERIAL	LOCATION
7075	ALL EXCEPT AS SHOWN BELOW
2024	LOWER SKIN LOWER STRINGER [illegible] CHORDS



SECTION B-B



must be rather narrow. A review of service and test failures indicates that a width of five inches might be appropriate. Again fortunately, research work has indicated that effects of chordwise defects in any of the chordwise elements are additive both from a residual strength and crack growth standpoint[1].

The element defined above is still not satisfactory because it is not homogenous. Typically, the lower cover is more critical in fatigue than the upper cover or the spar webs and typically the stresses increase as one transverses through the cross section from the front to the rear. A review of typical designs indicates that to obtain homogenity, top and bottom covers of the element defined above should be divided into quarters and each spar web should be treated separately. Thus, the old element would result in eleven new elements. The new elements would be redundant, but under the concept shown later in the sample logic diagram, this problem could be handled by considering all elements of the cross section to be "adjacent elements."

To this point, only chordwise defects in the wing have been considered. Spanwise defects, especially cracks, do occur in service but have little effect on the wing except on its torsion carrying capability and appear to be independent of chordwise defects. However, extremely long defects are necessary to significantly effect the torsional strength. (Cracks as long as six to ten feet have been experienced in service without

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[1] H. Laurence Snider, Franklin L. Reeder, and William Dirkin, Residual Strength and Crack Propagation Tests on C-130 Airplane Center Wings with Service-Imposed Fatigue Damage (Langley, Va.: National Aeronautics and Space Administration, 1972).

wing failure). Consequently, it may not be necessary to evaluate spanwise defects but if it is, spanwise and chordwise defects could be evaluated independently using corresponding spanwise and chordwise strips of the wing box. The same approach could be used on the fuselage where bending is the primary flight load failure mode and is affected primarily by circumferential defects whereas pressure loads are primarily affected by longitudinal defects. The above approach assumes that failure in one principal direction of stress are independent of failures in the other principal direction of stress and that failures in adjacent parallel elements are independent.

Sample Logic Diagram

Based on the foregoing discussions, the author developed a sample logic diagram which is described in detail and shown in Figure 9 in the Appendix. This diagram illustrates the life of a structural element as it progresses through development and production phases, enters service, and repetitively progresses through flight and post-flight phases. The element either develops defects or not in any one of these phases and the defects are either catastrophic or not and are either found or not found in post-flight inspections. The discovery of these defects either results in modifications of the element and/or inspection program or does not. The service portion of the logic diagram is repeated until the airplane crashes or is retired from service.

This element logic diagram could provide the basic building block for a computer program for evaluating the effectiveness of a given inspection program for a fleet of airplanes of a given design which is developed

and operated under a given set of policies. By use of computer simulation techniques, as shown in the flowchart of Figure 7, each element of each airplane could be run through the development and production phases of the diagram as each airplane enters service and then repetitively "flown" through the service phase of the logic diagram on a flight-by-flight basis until retirement or failure with the inspection program and/or structure modified as indicated by the logic diagram after each "element flight" and each "fleet flight" (i.e., a "fleet flight is completed when all elements of all airplanes in the service fleet have been "flown" through one flight). It is anticipated that the computer program would account for the fact that all airplanes of a given design do not enter service at the same time and that the entire fleet would be "flown" through the program on a flight-by-flight basis. That is, no element of any service airplane would embark on another flight until all elements of all service airplanes had completed the previous flight. This approach has the advantage in that it reflects reality wherein test and service experience on any airplane affects subsequent actions on all production and service airplanes. It is also anticipated that to simplify the program, all calendar flight times and fatigue lives would be expressed in terms of the number of flights that an average airplane would have made during the number of days or flight hours involved. This is based on the assumption that all airplanes will make the same number of flights per hour and per day.

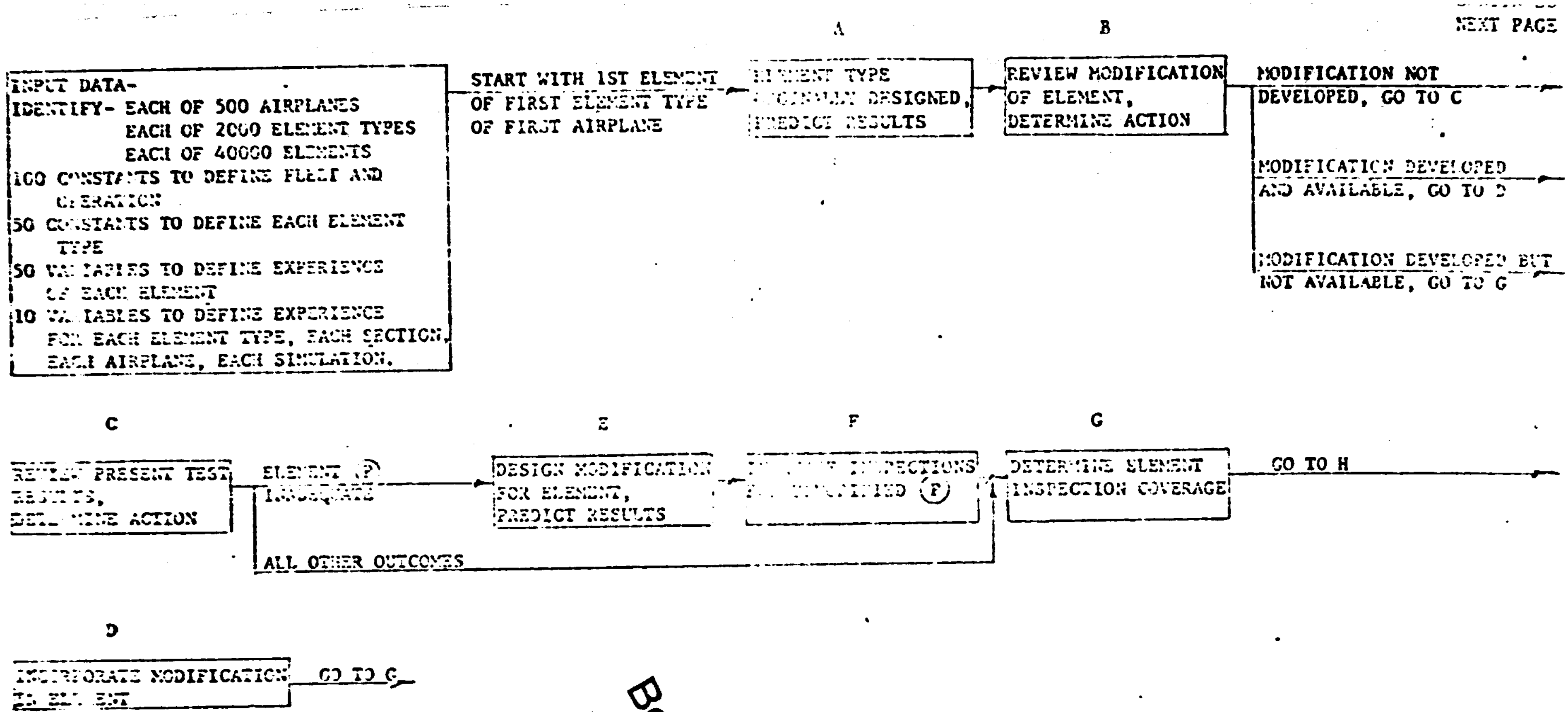
The expected results for a given situation could be obtained by repeated fleet trials of "flying" a fleet through the program from introduction of the design into service to retirement of all aircraft of the given

design with the results recorded for each trial and averaged for many trials to obtain the expected results.

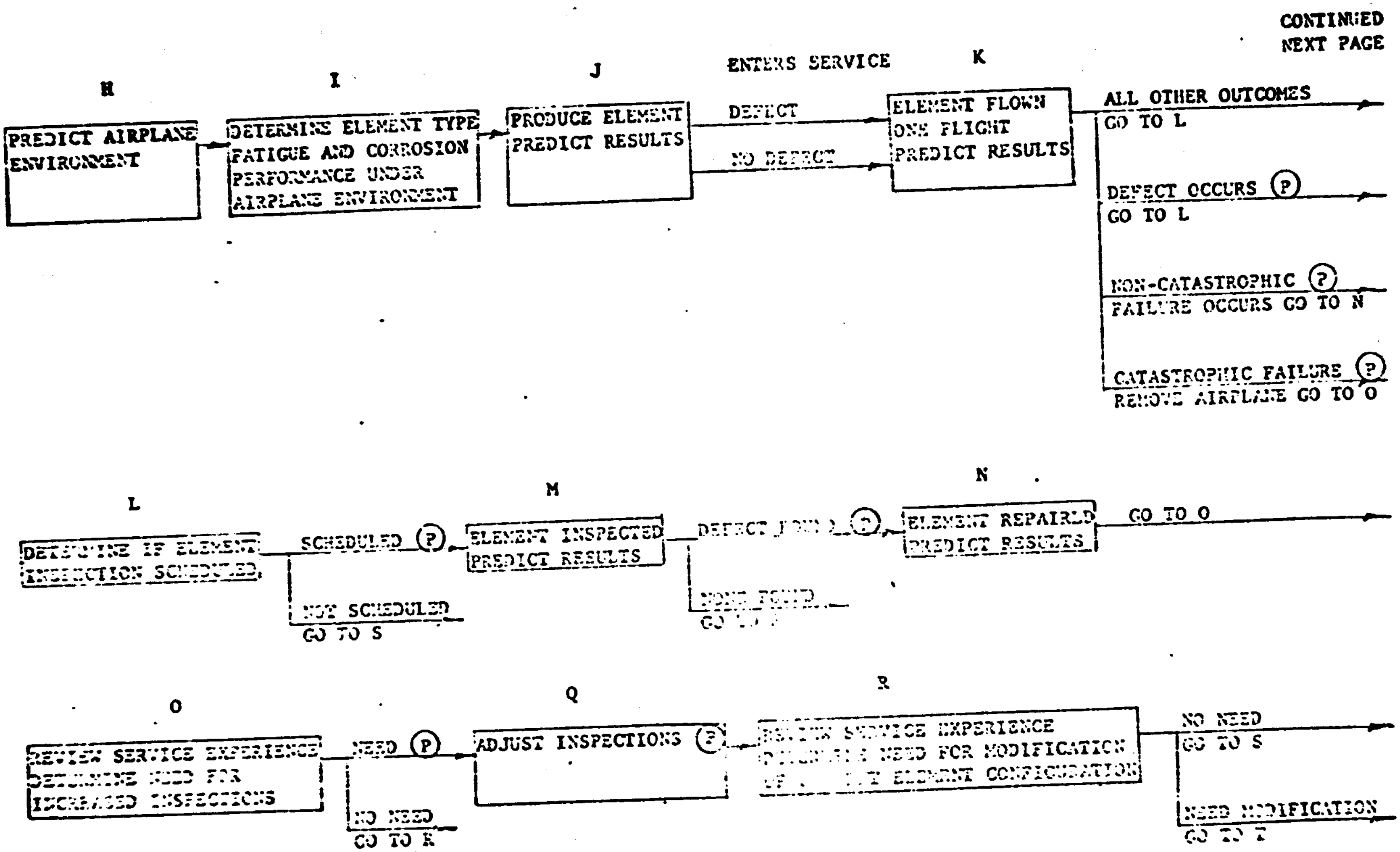
Figure 7

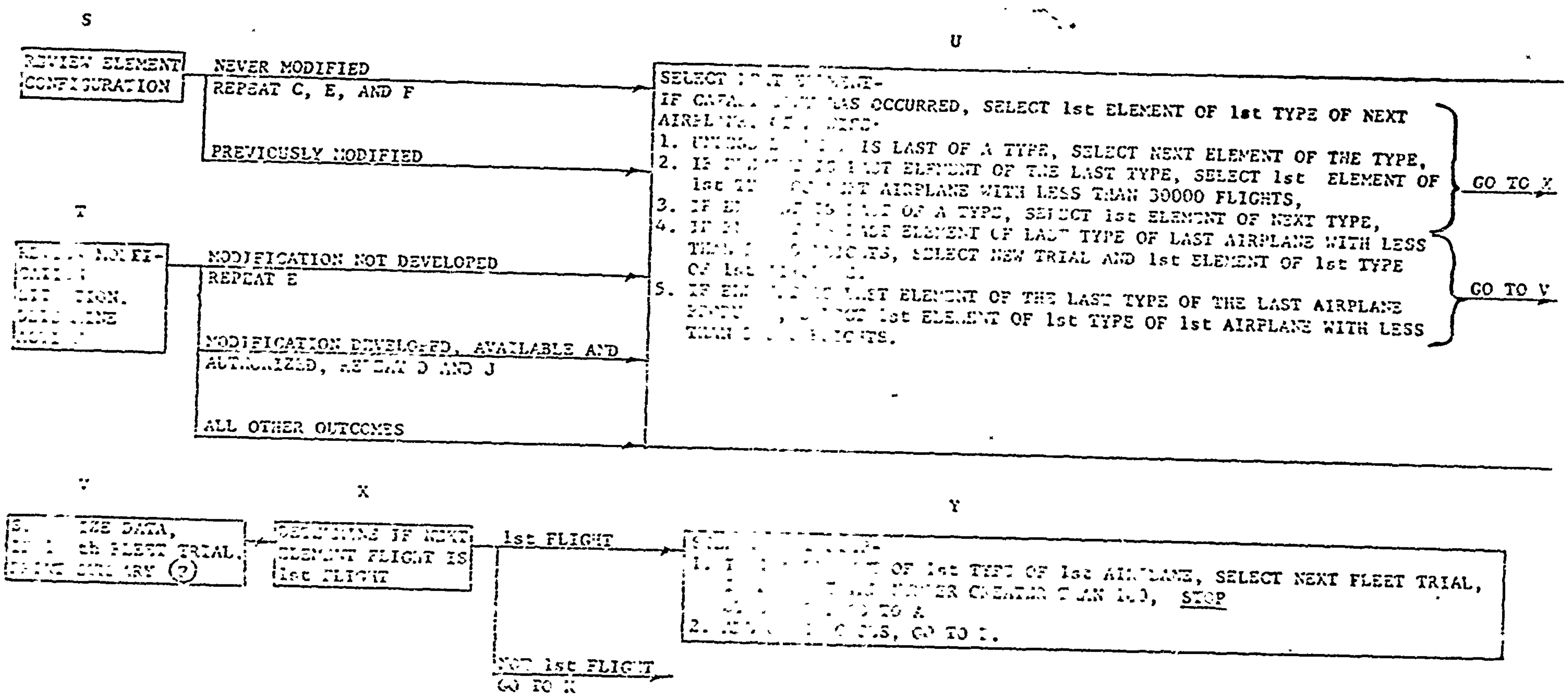
(3 Sheets)

Computer Simulation Flow Chart



NOTE: SYMBOL (P) INDICATES PRINT OUT OF ALL DETAILS OF INCIDENT


H
PREDICT AIRPLANE ENVIRONMENT
I
DETERMINE ELEMENT TYPE FATIGUE AND CORROSION PERFORMANCE UNDER AIRPLANE ENVIRONMENT
J
PRODUCE ELEMENT PREDICT RESULTS
ENTERS SERVICE
DEFECT
NO DEFECT
K
ELEMENT FLOWN ONE FLIGHT PREDICT RESULTS
ALL OTHER OUTCOMES GO TO L
DEFECT OCCURS P GO TO L
NON-CATASTROPHIC P FAILURE OCCURS GO TO N
CATASTROPHIC FAILURE P REMOVE AIRPLANE GO TO O
CONTINUED NEXT PAGE
L
DETERMINE IF ELEMENT INSPECTION SCHEDULED
SCHEDULED P
NOT SCHEDULED GO TO S
M
ELEMENT INSPECTED PREDICT RESULTS
N
ELEMENT REPAIRED PREDICT RESULTS
GO TO O
O
REVIEW SERVICE EXPERIENCE DETERMINE NEED FOR INCREASED INSPECTIONS
NEED P
NO NEED GO TO R
Q
ADJUST INSPECTIONS
R
NO NEED GO TO S
NEED MODIFICATION


S
REVIEW ELEMENT CONFIGURATION
NEVER MODIFIED
REPEAT C, E, AND F
PREVIOUSLY MODIFIED
T
MODIFICATION NOT DEVELOPED
REPEAT E
MODIFICATION DEVELOPED, AVAILABLE AND AUTHORIZED, REPEAT D AND J
ALL OTHER OUTCOMES
U
SELECT
IF HAS OCCURRED, SELECT 1st ELEMENT OF 1st TYPE OF NEXT
1. IS LAST OF A TYPE, SELECT NEXT ELEMENT OF THE TYPE,
2. LAST ELEMENT OF THE LAST TYPE, SELECT 1st ELEMENT OF AIRPLANE WITH LESS THAN 30000 FLIGHTS,
3. OF A TYPE, SELECT 1st ELEMENT OF NEXT TYPE,
4. ELEMENT OF LAST TYPE OF LAST AIRPLANE WITH LESS SELECT NEW TRIAL AND 1st ELEMENT OF 1st TYPE
5. LAST ELEMENT OF THE LAST TYPE OF THE LAST AIRPLANE 1st ELEMENT OF 1st TYPE OF 1st AIRPLANE WITH LESS
GO TO X
GO TO V
V
FLEET TRIAL.
X
ELEMENT FLIGHT IS 1st FLIGHT
1st FLIGHT
NOT 1st FLIGHT
Y
OF 1st TYPE OF 1st AIRPLANE, SELECT NEXT FLEET TRIAL,
STOP
TO A
GO TO

DEVELOPMENT OF METHOD FOR SYSTEM COST EVALUATION

Objective and Obstacles

As indicated in the definition of the problem, the ideal structural inspection program would be a minimum cost program in which all significant related costs are fully accounted for. Consequently, it is the objective of the system cost evaluation to identify all significant costs and within the constraints on the study, to estimate these cost for possible variations in the system. The major problem is to realistically estimate the costs for the system variations. The major obstacles are that a great deal of data is needed to determine the costs and the system is extremely complex. Fortunately, the system model described in the preceding chapter and in the Appendix can provide the necessary cost information as discussed below.

Significant Costs

As previsouly mentioned in the definition of the problem, as the inspection effort increases, defects are found before they become large and result in major repairs or accidents. Likewise, if modifications are incorporated, defects are prevented. Further if a full scale fatigue test of the entire airplane is performed, hidden deficiencies can be discovered prior to service, special inspections instituted, and modifications incorporated to prevent service defects. Consequently, it is apparent that inspection costs, repair costs, accident costs, modification costs and full scale fatigue test costs are significant costs related to the structural inspection program. Airplane down time costs are also a major cost, having been estimated to be 39,000 dollars a day for the Boeing 747. However, if

typical maintenance practices and situations are assumed, down time can be accounted for by applying an historical factor to inspection, modification and repair costs. In the case of planned activity such as inspections, the down time costs should be prorated between structural inspection and the other maintenance being concurrently performed.

Estimation of Individual Cost Components

Inspection and Repair Costs

The model described in the Appendix would accumulate the inspection and repair (including down time) costs for each inspection level as part of the determination in operation DS16 of whether a modification should be developed. By totalling these costs, the inspection and the repair costs can be determined for the complete life of the structure being evaluated. The cost input to the model can be obtained from historical data on past airplanes by multiplying the total cost for the percent of the total effort per visit applied to structural inspection or repair of a particular element. The resulting cost should be corrected to the size of the airplane being evaluated by the ratio of airplane empty weights. The maintenance costs for operational airplanes are published periodically in trade magazines such as "Aviation Week". The manhours for first and second level of inspection (which is the major expense) for the past narrow body jet transports were given in the section on "Inspection Frequency". Repairs made for defects found in the first level of inspection may result in flight cancellation and down time which should be added to the cost of the repair. A review of present jet transport experience[1] indicates that they average 2.6 days down time per

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[1] J. M. Wolgast, "Structural Character of the Jet Age." Proceedings of the 1968 ATA Engineering and Maintenance Conference, Phoenix, Ariz. (Washington, D. C.: Air Transport Association of America, 1968).

year for 2.5 such cancellations. Wolgast's review also indicated that for a Block Overhaul visit, which is essentially a third level inspection plus one-fourth of a fourth level inspection[1], the typical airplane required:

1250 manhours for structural inspection plus

942 dollars for outside services;

1933 manhours for structural repairs plus

1375 dollars for materials for 152 discrepancies; and

2050 manhours for structural modification plus

1375 dollars for materials.

Wolgast's review also shows that structural maintenance typically represents eight to ten percent of airline direct maintenance cost and accounts for twelve to fifteen days of down time per year of which a large part could be eliminated.if all structural maintenance were to be eliminated. It is believed that with these sources of information available, adequate estimates can be made of structural maintenance costs.

Modification Costs

. The model in the Appendix also estimates the modification cost in operation DS16 for each modification developed. This value could be accumulated to give total modification costs. The input data could be obtained in the same mannc. as for inspection and repair costs except that the cost of testing the mo.ification would have to be estimated from historical data. The model covers only fatigue modification costs.

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[1]Ansvitt, "Aircraft Structural Sampling Inspection Programs," p. [illegible].

Corrosion modification costs were not covered as no practical way could be found to estimate the unmodified corrosion defect occurrence rate. As a review of reported defects for the Boeing 720 indicates that approximately 25 fatigue defects occur for every corrosion defect, this lack of evaluation of corrosion modification policies should not be critical. The model also does not account for costs of modifications developed from the full scale fatigue test because different criteria would be used to decide whether such modifications should be developed. The model simulation would record such modifications and could be easily changed to account for their costs. Also the model does not account for the lower cost of incorporating modifications before rather than after delivery. Military experience indicates that a production modification costs approximately thirty percent of that of a modification made after delivery. Such a factor could be applied to production modifications.

Accident Costs

This study is limited to vital primary metal structure (fuselage shell, wing and empennage box beams, and fuselage attachments). Accident records indicate that failure of such structure is either catastrophic resulting in loss of airplane and occupants or resulting in a repair proportional to the size of the failure. Such failures rarely result in non-fatal accidents wherein the airplane is further damaged or destroyed. The costs of non-catastrophic failures is covered under repair costs. The model described in the Appendix will estimate the expected number of catastrophic failures for the lifetime of a fleet for a given situation. The accident costs can be obtained by multiplying the valuation of airplane and occupants

times this expected number. The value of the airplane can be easily estimated from original or depreciated cost. The cost of a Boeing 747 is approximately 20,000,000 dollars[1]. The economic value of human life is of course, a very emotional and controversial subject. However, a study of the subject[2] suggests a value on the order of 500,000 dollars for an airline passenger in 1975 based primarily on present value of expected future earnings.

Full Scale Fatigue Test Cost

A full scale fatigue test is not always performed, or continued to its maximum length, in a development program as it is extremely expensive. (The Boeing 747 fatigue test was estimated to cost 25,000,000 dollars for 60,000 accelerated flights.) Yet the existence and length of the full scale fatigue test effects structural maintenance costs. The cost of such a test can be estimated from a report by Harpur and Throughton[3] as a function of size of airplane and length of test.

Costs Not Considered

Neither the costs of corrosion modfications nor the cost of lost business because of a bad accident record have been considered. The first cost is not considered to be vital to optimization of structural inspection and the latter cost is considered to be too nebulous.

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[1] D. L. Greer and W. E. Knapton, Cost/Benefit Analysis of Incorporating Crash Resistant Fuel Tanks in a Civil Jet Transport Aircraft (San Diego, Calif.: Convair Division of General Dynamics, 1970).

[2] Gary Fromm, Economic Criteria for Federal Aviation Agency Expenditures (Cambridge, Mass.: United Research Corp., 1962).

[3] N. F. Harpur and A. J. Troughton, "The Value of Full Scale Fatigue Testing," Proceedings of International Committee on Aircraft Fatigue, Munich, 1965. (Oxford, England: Pergamon Press, 1965). Fig. 14.

SUMMARY

This paper presents a plan for FAA development of a method for establishing aircraft structural inspection programs on a rational basis and a justification for such development. As a plan must define the tasks necessary to achieve the objective, the definition of these tasks has naturally resulted in a preliminary development of the actual method.

The motivating force for this development is the belief that present structural inspection programs, which represent a massive effort, are far from optimum. To date these programs have been largely developed a judgemental basis from past experience with no attempt to use existing analytical techniques and data to estimate whether the resulting safety level would be adequate. The result has been a trend of ever-decreasing inspection effort. Service history was reviewed and it was found that seven fatal accidents involving transports of U. S. or British design, have occurred in the last seven years because the inspection did not detect structural defects. Further, laboratory testing has indicated that structural defects could initiate and grow to failure within some of the current inspection intervals. Conversely, inspection personnel were contacted and their reaction was that much of the present inspection effort is unnecessary. It was concluded that FAA development of a method for establishing structural inspection programs on a more rational, scientific basis than presently used might provide real benefits.

Rational development required a model which would permit an apriori evaluation of the effectiveness of an inspection program. The effectiveness is measured by the degree to which defects, resulting from various hazards, are detected prior to their causing accidents or major repairs. Recognizing the "garbage in, garbage out" problem as a major obstacle, considerable effort was made, even prior to developing a preliminary model, to assure that valid data could be obtained from available sources for the input required by the model.

A preliminary scenario of the model identified the major input data requirements as - (1) defect occurrence rates for each hazard, (2) strength degradation rates for fatigue and corrosion defects, and (3) inspection frequency and reliability. A search of the literature and service experience identified initial fatigue, corrosion, birth defects, and service damages as the primary hazards which could be minimized by inspection. The available sources for input data were evaluated. It was concluded that items (1) and (2) above could be obtained from a review of airline mechanical reliability reports, the manufacturers design data, and the literature on structure and that inspection reliability estimates could be obtained from a survey of maintenance experts.

The modeling approach selected was of using computer simulation to evaluate a relatively complex model covering a major portion of the primary structure of the airplane and accounting for most of the major variables of service usage and inspection. This approach was selected over using a rigorous mathematical analysis to evaluate a simple, idealized model because it was considered to be more realistic and more easily understood by the layman.

A great deal of effort was expended in the definition of the planned computer simulation by the simulation flow chart of Figure 7 and the logic diagram in the Appendix to assure that the simulation would be reasonably valid and realistic. A procedure for dividing the structure into elements was outlined. The logic diagram illustrates the life of such an element as it progresses through developmental and production into service and repetitively through flight and post-flight phases with defects occurring, being detected or resulting in failure thus causing changes in inspection and/or structure. The logic diagram and flow chart defining the simulation were submitted to Control Data Corporation who advised that the approach was feasible from a computing standpoint and provided a cost estimate for the simulation.

Rational development of an inspection program also required an apriori evaluation of the associated inspection, repair, modification, and accident costs. An approach was outlined for estimating these costs from the information generated in the computer simulation of the inspection effectiveness model. Accident costs were based on aircraft losses plus the estimated future earnings of the occupants. Minimization of the above costs was selected as the optimization criteria. At this point a major conclusion of the paper was reached - that a valid rational method of establishing an airplane structural inspection program was feasible.

To determine whether FAA development of the rational method was desirable, estimates were made of the benefits and of the costs of development and application. The cost of development and application to a hypothetical fleet was estimated to be $316,000. The benefits were predicted from

the changes in the present accident rates and inspection intervals indicated by previous studies of the problem primarily by Eggwertz[1]. The net benefit, after deduction of increased inspection costs, was estimated to be equivalent to $20,000,000 for the hypothetical fleet. This resulted in the second major conclusion of the paper - that development and application of the rational method would pay for itself many times over and therefore FAA development was desirable.

The tasks, schedules, and required manpower were then defined for FAA development of the rational method and the demonstration to industry. It was concluded that outside assistance would be necessary to develop and run the computer simulation and that presentations of the results at major technical and trade meetings would provide an appropriate demonstration.

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[1]Eggwerts, Study of Inspection Intervals for Fail-Safe Structures.

APPENDIX

LOGIC DIAGRAM AND SYMBOLOGY

The logic diagram outlined in the body of this study is further defined in this Appendix by showing the diagram in more detail in Figure 9 and by describing the functional relationships and their basis.

As original symbology was used in the logic diagram, it will be described. Each horizontal line segment between intersections with circles and/or vertical lines, indicates a state, a range of states or an event. A chain of these horizontal line segments gives the life history of a given element in a given airplane. Each circle indicates a functional relationship which determines what event or state will follow. The letters in the circle indicates the type of function as follows:

First Letter –

C – indicates that a simple constant, which does not change throughout the simulation, is applied.

D – indicates that functional relationship is deterministic.

P – indicates that functional relationship is probabilistic.

Second Letter –

D – indicates that functional relationship is dependent only on information generated in the developmental or production phases.

S – indicates that functional relationship is dependent on information generated in the service phase.

The numbers in the circles identify the particular functional relationship. The vertical lines indicate that the logic diagram that follows applies to each of the states or events whose horizontal line is intersected. The arrows indicate that the state or event which it points to, automatically follows.

It is suggested that the multiple sheets of the logic diagram be removed from the report and taped together to make the diagram easier to follow.

Functional Relationships

The functional relationships identified on sample logic diagrams are defined in the following paragraphs which are identified by the functional relationship symbol. The X symbols and associated numbers identify variables which may change during simulation of a given situation.

C1 - "The predicated average fatigue life of element by analysis" is a constant number of average flights which can be obtained from a summary of the fatigue analysis[1] for the pertinent area of structure. It must be corrected for the size effect involved in going from the test specimen on which the analysis is based to a structure the size of the element[2]. This correction is based on the concept that the probability of a fatigue crack not initiating in a large structure is equal to the

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[1]Such as given in Boeing Co. brochure, Boeing 747 Fatigue Integrity Program, Document No. D6 - 13050-773 (Seattle, Wash.: Boeing Co., 1970).

[2]Sikorsky Aircraft, Fatigue Properties and Analysis, p. 36.

product of the probabilities of each part not initiating a crack. On this basis, the probability of failure of a structure four times the size of another structure is approximately four times greater, all other things being equal. With the relative probabilities known, the correction may be made by obtaining a fatigue life distribution (typically log-normal)[1] and a standard deviation from the literature and interjecting the relative failure probabilities.

FDI - As fatigue analysis is notoriously inaccurate, the actual average fatigue life of elements of a given design (X2) will usually be different than predicted by analysis. The probability of the actual life being a given percentage greater or less than predicted could be obtained from a British study of the subject[2]. By plotting this probability as a cumulative frequency curve and by use of Monte Carlo computer techniques[3] the percentage correction factor (X1) can be established for a given fleet trial. The value of X1 would change with each fleet trial but would have to be calculated only once for the first element to enter service on each fleet trial and then only once for all elements (regardless of location in the airplane) of the same design with the same general loading. The same cumulative frequency curve can be used for all element design.

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[1] G. E. Dieter and R. F. Mehl, Investigations of the Statistical Nature of the Fatigue of Metals, Technical Note [illegible] (Washington, D. C.: National Advisory Committee for Aeronautics).

[2] Raithby, A Comparison of Predicted and Achieved Fatigue Lives

[3] Maurice Sasieni, Arthur Yaspan, and Laurence Friedman, Operations Research - Methods and Problems (New York: John Wiley & Sons, Inc., 1959), p. 59.

DS1 - This functional relationship is a simple variable (X3) which indicates for a given fleet trial whether any previous decision has been made to develop a fatigue modification for a given element design in a given airplane location. The logic diagram assumes that if a previous servi. fatigue failure has instigated a design modification, a fatigue test failure will not instigate an additional modification.

DD1 - This function for determining whether the airplane fatigue test has exceeded the average design fatigue life of an element before a given element enters service is a direct arithmetic relationship to:

(X4) the production number of the airplane being evaluated (entering service).

(C2) the number of airplanes produced per unit of time.

(X5) the time since first airplane entered service.

(X6) the time between (plus or minus) first airplane delivery and start of the fatigue test (would be infinite if never started).

(C3) the number of accelerated flights on fatigue test per unit of time.

(C4) the acceleration factor on fatigue test.

(X2) the actual average fatigue life of element design; and

(C5) maximum number of accelerated flights to be imposed in fatigue test.

The logic diagram assumes that a fatigue test failure of the element will occur when the number of simulated flights on the fatigue test times C4 exceeds X2. It is also assumed that production test rates are constant.

DD2 - It is assumed that decision whether or not to install a modification o. a given element in a given location will depend only on whether the average fatigue life of the unmodified design (X2), as reflected by the test failure, exceeds a pre-selected level (C6).

DD3 - This functional relationship reflects the policy assumed for the given situation. Experience on the Boeing 747 indicates[1] that the policy . be adequately defined by the percentage (C7) of test failure life (X2) at which increased inspections become effective and by the factor (C8) by which the close external inspection frequency is increased and by the factor (C9) by which the sample size of internal inspections would be increased.

C10 - This is a constant which reflects the policy assumed on whether or not modifications are to be tested before installation. The logic diagram assumes that if tested, the modification will have a longer lead time but the average modified fatigue life (X7) will be the same as predicted for the original design. However, if not tested, there is a possibility of the fatigue life being different than predicted.

DS2 - Whether modification is available for production incorporation is a direct arithmetic function of:

(X8) the time between first airplane delivery and the decision to develop modification;

(C11) the lead time to produce modifications;

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[1]Boeing Co., 747 Structural Inspection Program Revision, Feb. 2-3 (Seattle, Wash.: Boeing Co., 1972).

(C12) the lead time to incorporate modification in production; and

(X4) and (C2) which, can be used to determine time between delivery of this airplane and the first airplane delivered.

The logic diagram assumes that if a modification has been developed for a given element and it is available for production incorporation before delivery, it will be installed. It also assumes that a modification is not available at the time the decision is made to develop a modification.

DD4 - The inspection coverage of each element will be defined when the first airplane of a given design enters service and would not have to be re-defined for any subsequent simulations. It may be desirable to perform DD4 through C17 manually for each element prior to computer simulation.

C13 thru C17 - These are constants which define the structural element and inspection coverage as indicated and can be obtained from inspection criticality analysis (See Figure 1). The inspection coverage indicated by the logic diagram is based on previous discussions in the body of this report under "Inspection Frequency" and "Inspection Reliability". The logic diagram assumes that there is no significant chance of detecting defects in inspection which do not cover defect areas of fatigue life.

PD2 - The service loading environment varies considerably between individual airplanes[1]. The purpose of this functional relationship is to establish

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[1]Anderjaska, Fatigue Substantiation Procedures, p. 2.

a correction factor (X9) to multiply times the actual average fatigue life (X2 or X7) for a mean load environment to obtain the actual average fatigue life of the element for the loading environment of a given airplane. By assuming a distribution for the variation of the rate of flight load occurrence between airplanes (typically log-normal) and a standard deviation for the distribution, the ratio of the rate of load occurrence to the mean can be obtained for an individual airplane by the previously discussed Monte Carlo methods. The inverse of this ratio to the mean will equal X9. A separate correction factor (X10) would have to be obtained by similar means for elements loaded by cabin pressure loads. It should be noted that the average fatigue life for an individual element with the latest modification (X7) and the average fatigue life for an individual element with the previous (or no) modification (X2) varies as the element goes through its life, experiences a load environment different from the mean, receives production and/or service damage, corrodes, is repaired and so on.

DD5 - The predicted crack growth rate for the type of construction and material and a test loading environment can be obtained from manufacturers test data (such as shown in Figure 3) as discussed under "Rate of Strength Degradation Due to Fatigue". It is anticipated that this variable rate can be approximated by two constant rates, one representing the earlier slower crack growth period (C18) and a higher rate (C19) starting when the crack reaches the critical crack length (C20).

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[1] Dieter, Investigation of the Statistical Nature of Fatigue of Metals.

DD6 - The predicted fatigue crack growth rate for the element can be obtained by multiplying the crack growth rates of DD5 by correction factors to account for differences between the load environment of the particular element in the particular airplane and the test and for differences between the corrosive environment of the test and the element in the airplane. The correction factor for the load environment would be the inverse of the product of X9 or X10 and the ratio (X11) of the test rate of load occurrence to the mean service rate of load occurrence for a given element. The severity of the corrosion environment of the element in the airplane is defined the corrosion rating factor in the manufacturer's Inspection Criticality Analysis Figure 1. The multiplying factor (C21) by which the corrosive environment (represented by the rating) accelerates fatigue crack growth can be obtained from generalized fatigue tests under such environment[1]. The critical crack length (C20) of DD5 should be expressed in terms of reduction of strength (in per cent of design limit load) that would be caused by a crack of that length in the section defined by the element and "adjacent" elements. To be most useful the crack growth rates should also be in these terms.

DD7 - The corrosion degradation rate, as previously discussed under this general heading, could be obtained from well documented cases of corrosion in service and expressed as straight line functions of the corrosion resistance rating of the element (C22) and the time to initiate corrosion (X12). The generalized degradation rate should be expressed in terms of cross sectional area lost per flight. This

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[1]J. Schijve, Cumulative Damage Problems in Aircraft Structures and Materials (Amsterdam, Holland: National Aerospace Laboratory, 1969), p. 1.1/25.

functional relationship could hold for all elements in the airplane and with it the corrosion degradation rate established for the element as function of time to initiate corrosion.

DD8 - This function changes the corrosion degradation rate function of DD7 a strength degradation rate by multiplying the rate by a factor representing the strength (in per cent of design limit load) lost in the section (defined by the element and "adjacent" elements) per unit of cross sectional area lost. This function can be defined as a straight line function of X12 by slope (C23) and intercept (C24).

PD3 - The probability that an element will enter service with a production defect involves an "either or" or binomial probability distribution. The probability of there being a defect (C25) in a given element can be obtained from the "Birth Defect Occurrence Rate" which can be obtained from service experience as discussed under that heading. The outcome for a given element on a given airplane in a given computer trial can be determined by Monte Carlo techniques as discussed under PD1.

DD9 - This functional relationship involves reducing the average fatigue life of an element with a production defect by a constant factor (C26). This factor, as discussed under "Birth Defect Occurrence Rate" could be obtained by reviewing service experience to identify the most prevalent type of birth defect and then reviewing fatigue test literature to determine what is the effect on fatigue life. It is anticipated that the same factor would apply to all elements in the airplane unless service experience indicated that some other type of damage was most prevalent in a particular type of structure.

DD10 - This function for determining whether a given airplane enters service is a direct relationship to (X4) the production number of the airplane, (C2) the production rate and (X5) the time since the first airplane entered service.

DS3 - This functional relationship is a simple variable (X12) which indicates whether a corrosion defect has previously occurred in the element and the flight time on the airplane when the defect initiated and whether it was initially internal or external.

PD4 - This is a probabilistic function which is based on an exponential probability density function. As previously discussed, the Monte Carlo methods can be used to determine whether a corrosion defect occurs in the element on a given flight. The rate of occurrence (X13) can be obtained from service experience as discussed under "Input Data" and is anticipated to increase linearily with time since first airplane delivery (X5)

PD5 - This is a probabilistic function which is based on an "either, or" binomial probability density function, with Monte Carlo methods used to determine whether corrosion initiates in a stress concentration given that corrosion is initiated on a given flight. The probability of corrosion occurring in a stress concentration (C27) could be obtained by analysis of service corrosion cases for the type of structure involved. The type of structure could be categorized by the corrosion resistance rating.

DD11 and DD12 - In these operations, the actual average fatigue life of the element (X2 or X7 as applicable) is reduced by multiplying factors (C28) and (C29) for corrosion in a stress concentration and not in a stress concentration respectively. This reduction factor can be obtained from generalized fatigue tests under corrosive conditions. Such tests[1,2,3] indicate that a life reduction factor on the order of .15 to .25 for C28 and .50 for C29 might be appropriate.

PD6 - This is the same type of functional relationship as PD5 but involves the conditional, probability of corrosion initiating in an external part (C30) given that corrosion is initiated on a given flight. This probability can also be obtained from service experience for the type and location of structure involved. If corrosion has previously occurred, this outcome was previously determined.

DS4 - In this operation, the previous strength reduction of the section (X14) due to this defect is increased by the strength degradation due to corrosion in one flight which is determined by entering X12 (the time for corrosion to initiate in the element) into the degradation function determined in DD8. The present strength of the section (X15) also reduced by the amount corrosion grew in the flight. X15 was originally

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[1]Naval Air Engineering Center, Corrosion and Fatigue Evaluation of Spar Cap Specimens.

[2]Rosenfeld, Effects of Corrosion on Fatigue Life.

[3]Leybold, Investigation of the Effects of Atmospheric Corrosion.

the virgin strength of the section (C31). It should be noted that X has been reduced by previous defects in the elements and all other elements in the section. The logic diagram assumes that only one corrosion defect will be present in the section at a time.

DS5 - Whether internal corrosion becomes external depends on whether X1' exceeds a previous selected level (C32) which is based on the amount of material that must be eaten away for corrosion to become external.

PD7 - This functional relationship is the same as PD4 except it applies to service damage in lieu of corrosion and the rate of occurrence (C33) is a constant which can be obtained from service experience. The logic diagram assumes that the only affect a service damage or production defect has is to reduce fatigue life.

DD13 - In this operation, the actual average fatigue life of the element (X2 or X7 as applicable) is reduced by a multiplying factor (C34) which can be obtained from the results of generalized fatigue tests with and without typical service defects.

DS6 - This functional relationship involves evaluating a simple variable (X15) which indicates whether a fatigue crack has previously occurred in the element, the flight time on the airplane when the defect initiated and whether the defect was initially internal or external. Additional variables (X16 through X18) could be provided for the situation when multiple fatigue cracks occurred in a given element.

DS7 - This function involves further evaluation of X16 through X19 to determine whether cracks are internal or external.

DS8 - In this operation, the previous strength reductions (X20 through X22) due to cracks (X16 through X19) are increased to the crack growth in one flight. The crack growth rate (expressed in rate of strength reduction) is obtained from DD6 by determining whether the crack length (X20 through X22) exceeds the critical length (C21) above which the high crack growth rate applies.

DS9 - This operation is the same as DS5 except it is applied to defects X20 through X22 where applicable.

PD8 - This is a probabilistic function which is based on a log-normal probability density function whose variable is element fatigue life with a mean equal to the actual average fatigue life (X2 or X7 is applicable) and whose standard deviation (C35) from literature is on the order of .15[1]. The probability of a classical fatigue crack occurring on a given flight is equal to the probability of the element having a fatigue life equal to the number of flights accumulated at the time of the given flight. This probability can be obtained from the probability density function. The accumulated number of flights can be obtained from (X5) the time since first airplane delivery, (X4) the production number of the airplane being evaluated, and (C2) the number of airplanes being produced per unit time. With the above

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[1]Anderjaska, Fatigue Substantiation Procedures.

probability determined for a given flight, the outcome for that flight can be determined by inserting this probability into a binomial distribution and evaluating by Monte Carlo techniques. The logic diagram assumes that there is a possibility of a classical fatigue crack occurring in an element on every flight even though one has previously occurred.

PD9 - This is the same type of functional relationship as PD6 except it involves fatigue cracking instead of corrosion and has a different conditional probability (C36).

PD10 - This functional relationship is the same as PD4 except it applies to non-classical fatigue cracks in lieu of corrosion and has a different rate of occurrence (X23).

DS-10 - This operation is a simple determination of whether X12, or X16 through X19 indicate the presence of a detectable corrosion or fatigue defect. The logic diagram does not provide for evaluation of the probability of structural failure with no defect present. Production defects and service damage defects are assumed to be initially unimportant from a strength reduction standpoint.

PD11 - This function involves a probabilistic determination of the maximum flight load and cabin pressure experienced by the airplane in a given flight and a comparison of this load with (X15) the strength of the section to determine whether failure occurs. The cumulative distribution function for the probability of various maximum load levels can be expressed by a curve for flight loads and a curve for cabin

pressure loads. As discussed under "Rate of Load Occurrence" these curves can be obtained from research literature. The maximum flight and pressure load for a given airplane on a given flight can be obtained by Monte Carlo methods from the cumulative distribution functi... This determination would have to be made only once for each airpla... for each flight.

PD12 - This functional relationship involves the conditional probability of the failure either being catastrophic or not given that a failure of the section occurs. The conditional probability of catastrophy for massive failures of flight (C37) and pressure structure (C38) can be obtained from past experience. The outcome for a particular failure can be obtained by Monte Carlo Methods by using these constants in a binomial distribution. These constants would hold for all elements.

DS11 - This operation is a simple determination of whether (X7) indicates that the element has been previously modified.

DS12 - Whether an inspection is scheduled or not is a direct function of whether (X5) the time since first airplane delivery is a multiple of (C39) the first level inspection internal (cursory walk around) and whether (X24) indicates that an immediate inspection of this element was specified as a result of discovery of a defect on the previous flight.

DS13 - Whether the inspection covers the defect area is a function of whether the defect is now external or internal and whether the level of inspection scheduled covers the element and the defect location. The

level of inspection scheduled can be determined from (X..), (X4) t.. production number of the airplane, (X5) time since first airplane delivery, and the inspection intervals for the various levels of inspection in terms of multiples of (C39) - (C40, second level or area inspection), (C41, third level or close external inspection), (C.., fourth level or internal inspection) and (C43, fourth level inspec... sample).

PS1 - Whether a defect is found by the inspection which covers the defect area involves probabilistic "either, or" binomial function, from which the outcome for a given defect and a given inspection can be determined by Monte Carlo methods if the probability of detection is known. The probability of detection is considered to be a function of the level of inspection, the type of defect (fatigue crack or corrosion), the size of defect (X14, X20 through X22) the general type of structure (wing, fuselage, or fittings), and whether defect would involve fuel leak (C14). It is anticipated, as discussed under "Inspection Reliability," that discrete values of probability of detection would be obtained through a survey of maintenance experts for discrete situatio.. defined by the above variables. The logic diagram assumes that production defects and service damage are never found unless they have initiated with fatigue cracks and that when such cracks are found, they are repaired prior to further flight with the repair removing all effects of the production defect or service damage. It also assumes that repairs restore element (except for other defects present) to the

original predicted life and remove the strength degradation in the section due to the defect that was repaired. It is assumed the repairs are not fatigue tested.

DS14 - Whether a special immediate fleet inspection of the given element is specified as a result of finding a defect depends on whether the defect indicates that it is too hazardous to depend on scheduled inspections. A large defect indicates that the inspection system is not effective and a defect with an inspection interval large with respect to strength degradation rate, also indicates a hazardous situation. Consequently, criteria for specifying an immediate inspection can be expressed in terms limits of strength reduction (C44) for initial defects and ratio of inspection interval (lowest effective level) to crack growth (C46) or strength reduction (C45) for subsequent defects not found in special inspections. With the above constants applying to structure supporting flight loads additional constants are needed for cabin pressure loaded structure (C47), (C48), (C49). These constants would apply to all elements in all airplanes in a given simulation. The strength degradation rate would be based on (C18), (C19) and the information used in DD7 and the inspection interval would be the lowest level of close inspection of the defect area.

DD15 - Whether the frequency of the lowest level of close inspection of the element is increased as a result of finding a defect depends on whether the probability of detection needs to be increased to provide adequate safety. Such an increase is indicated when large and/or

numerous defects have been found in a given configuration of the element with a defect is long with respect to the rate of strength degradation. Consequently, criteria could be specified as requiring an increase in the close inspection frequency of element by a multiplying factor (C50) (or some portion of the strength degradation time) when:

1. The strength reduction of the defect found exceeds (C51), or
2. The strength reduction of the defect found exceeds (C52) and the ratio of the lowest level close inspection interval to the strength degradation rate exceeds (C53), or
3. The defect is found in a special immediate inspection and the ratio of the lowest level close inspection interval to strength degradation fate exceeds (C53), or
4. The cumulative strength degradation of defects found (including failures) in a given element (X25) of one configuration in all airplanes exceeds (C54).

The above constants apply to structure supporting flight loads, additional constants (C55), (C56), (C57) and (C58) are needed for cabin pressure loaded structure. The inspection program changes of DD15 and DD14 apply to external inspections if the defect extends to external areas, otherwise they apply to internal inspections. Some element or the number of minimum time between inspection frequency changes to prevent excessive changes.

DS16 - Whether a decision is made to develop a retrofit fatigue modification for delivered airplanes as a result of finding a service fatigue

defect depends on whether the cost of repair per flight plus the cost of added inspections per flight exceeds the cost of modification per flight. The cost of repairs per airplane flight can be estimated by accumulating the cost of repairs for each type of defect (fatigue (X26) or corrosion (X27)) and dividing by the accumulated number of flights. The cost of each repair could be estimated from the weight or volume of material added by multiplying cost per volume (C59) times the length, width and thickness of repair. The dimensions of the repair would be a function of average thickness of the element (C60), the reduction of strength due to the defect (X14 or X16 through X19), whether the reduction in strength exceeds some value (C61) which would require complete replacement for heavy structures. The actual cost of repair would have to be multiplied by a downtime factor (C62) and a factor (C63) which would be the airline's acknowledgement that defects may cause accidents. This later factor would be used only in deciding whether to develop a modification and would not be included in accumulated repair costs. (C59) could be initially based on production experience but should be normalized so that simulation of a typical situation will result in average cost per repair that agrees with service experience.

The cost per element inspection for each level of inspection can be estimated by dividing the cost of inspecting the entire airplane (including down time) among the elements, then the cost for each level of inspection can be accumulated (X28, first level), (X29, second level), (X30, third level), (X31, fourth level) and (X32, special inspections).

The cost of the added inspections per airplane flight can be determined by multiplying the cost per element inspection (C64) or (C65) (for the third or fourth level inspection involved) times (X33) or (X34) the accumulated inspection frequency increases due to application of (C50) to third or fourth level inspections. Dividing this result by the present inspection interval would give added costs per airplane flight. The logic diagram assumes that modification will reduce inspection frequency to original values. The cost of each modification could be estimated on the same basis as repairs (cost per volume of material added). The volume added for a typical case could be expressed in terms of percent (C66) of material in the element. The product of (C59, cost per volume) (C60, average thickness) (C67 width of element), (C62, downtime factor) and (C66) would give the cost per modification. This value plus the prorated test cost divided by the average number of flights remaining before retirement would give the modification cost per flight. The average number of flights remaining would be a function of (C68) number of airplanes in the fleet, (C69) the airplane service life, modification lead time (C11 plus one-half of the inspection interval of the level of inspection at which modification is incorporated).

The logic diagram assumes that modifications are not made for corrosion defects. This is not unrealistic as the corrosion defect occurrence rate is based on service experience which includes the effect of a typical modification program.

DS17 - Whether the modification is available for incorporation is a direct function of (X8) the time between first airplane delivery and the decision to develop the modification, (C11) the lead time to produce the modification and the time since decision to develop the modification (function of X4 and C2).

DS18 - Whether the modification is installed or not depends on whether the level of inspection being made equals or exceeds the lowest inspection level at which modifications are planned (C70) and whether the flight time on the airplane exceeds the (C71) percentage of the lowest flight time at which a fatigue defect found in test or service (X35).

DS19 - Same as DS1 except it applies to additional fatigue modification.

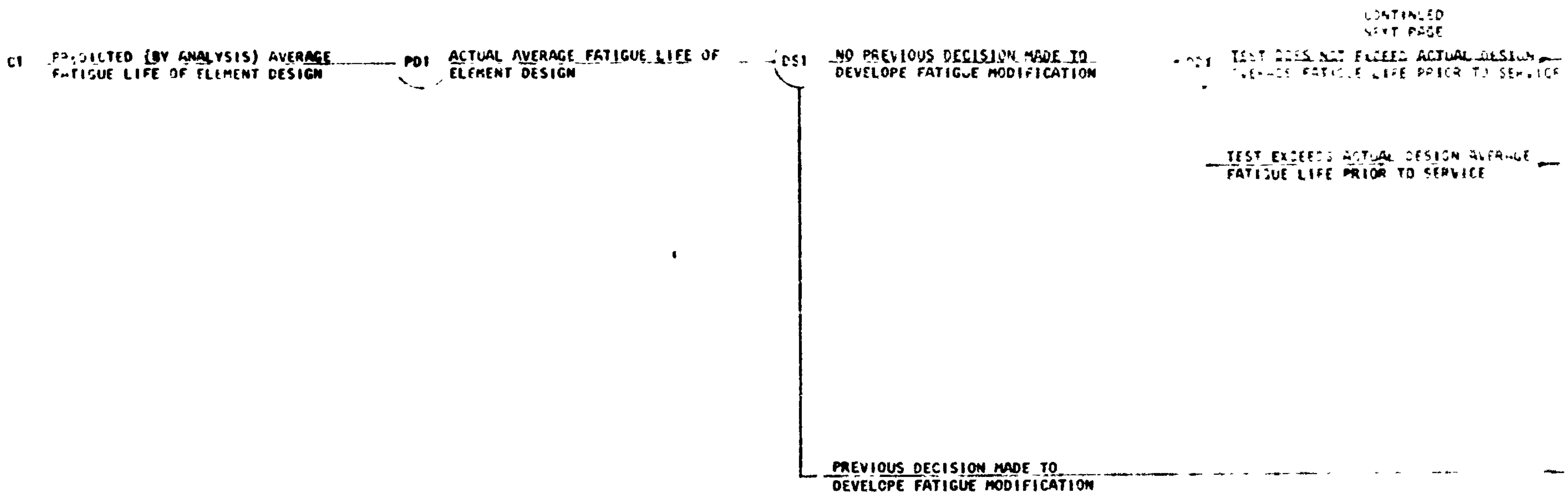
DS20 - It is assumed that no additional modifications are developed until the previous modification is installed on all aircraft.

DS21, DS22, DS23 - ame as DS16 through DS18 except they apply to additional modifications

Figure 9

(13 Sheets)

Sample Logic Diagram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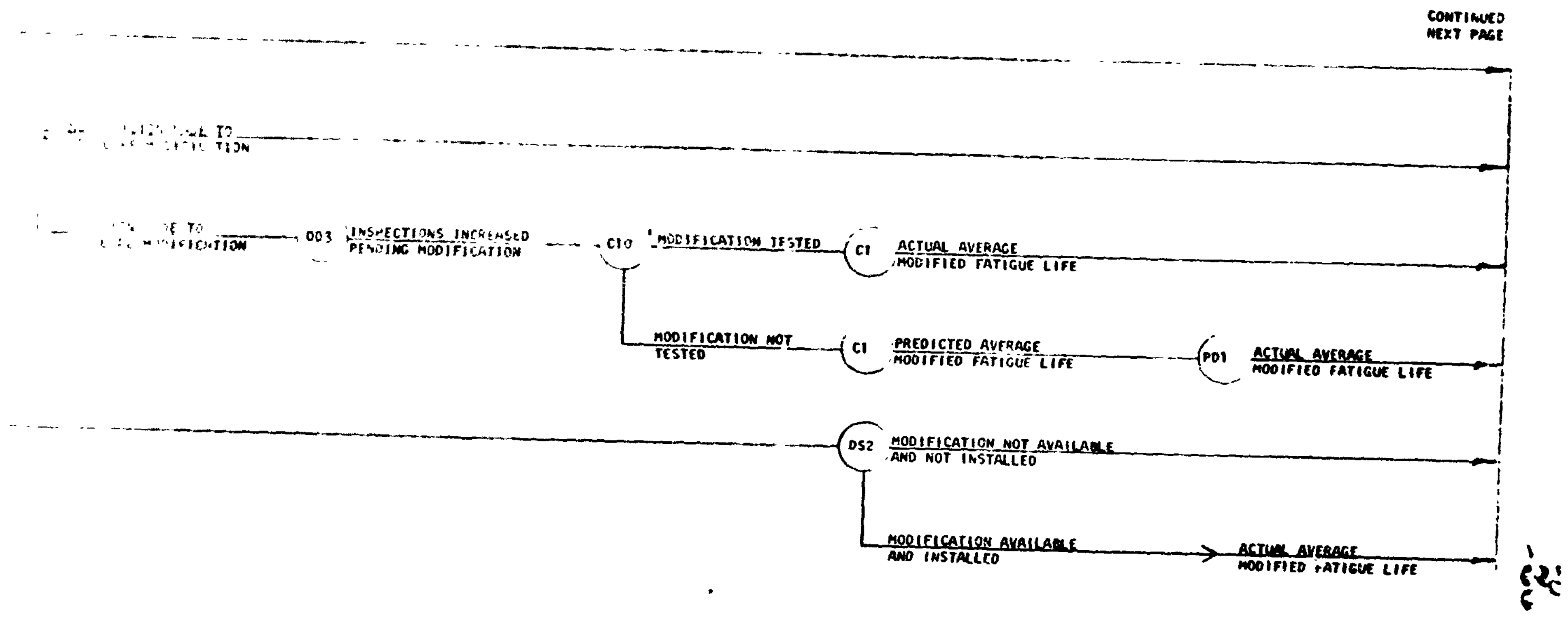
NEXT PAGE
C1
PREDICTED (BY ANALYSIS) AVERAGE FATIGUE LIFE OF ELEMENT DESIGN
PD1
ACTUAL AVERAGE FATIGUE LIFE OF ELEMENT DESIGN
DS1
NO PREVIOUS DECISION MADE TO DEVELOPE FATIGUE MODIFICATION
TEST DOES NOT EXCEED ACTUAL DESIGN AVERAGE FATIGUE LIFE PRIOR TO SERVICE
TEST EXCEEDS ACTUAL DESIGN AVERAGE FATIGUE LIFE PRIOR TO SERVICE
PREVIOUS DECISION MADE TO DEVELOPE FATIGUE MODIFICATION



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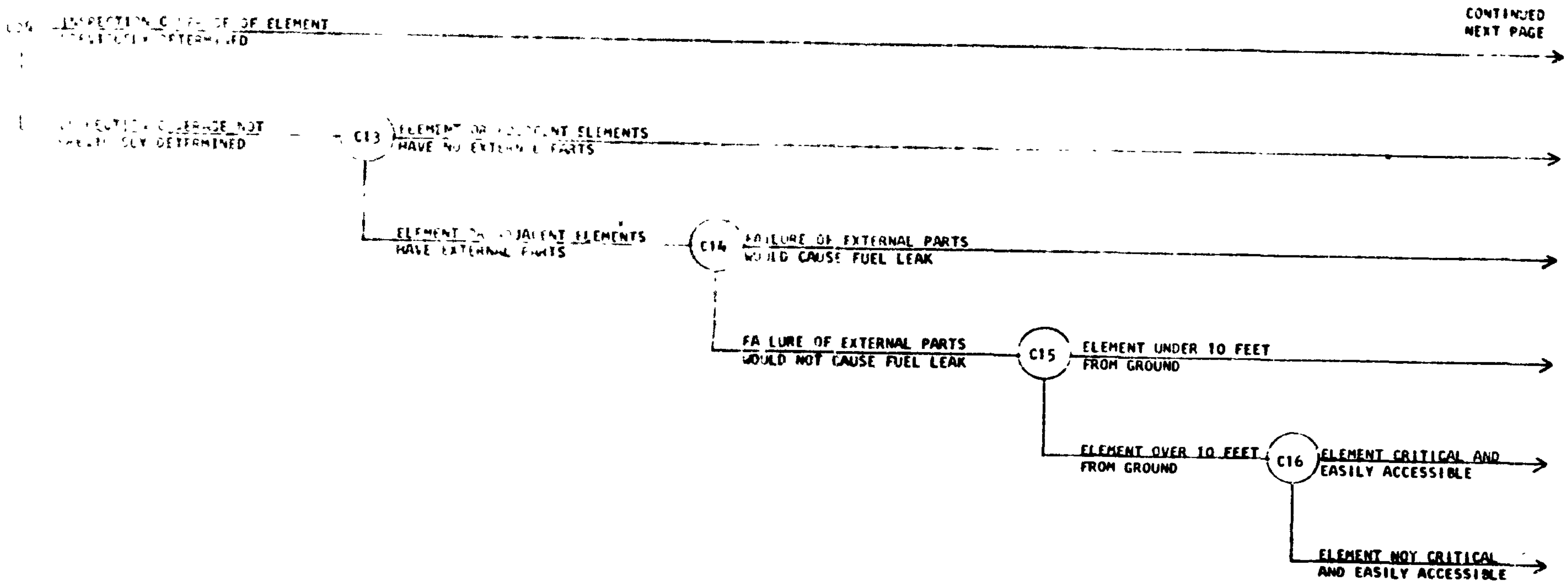
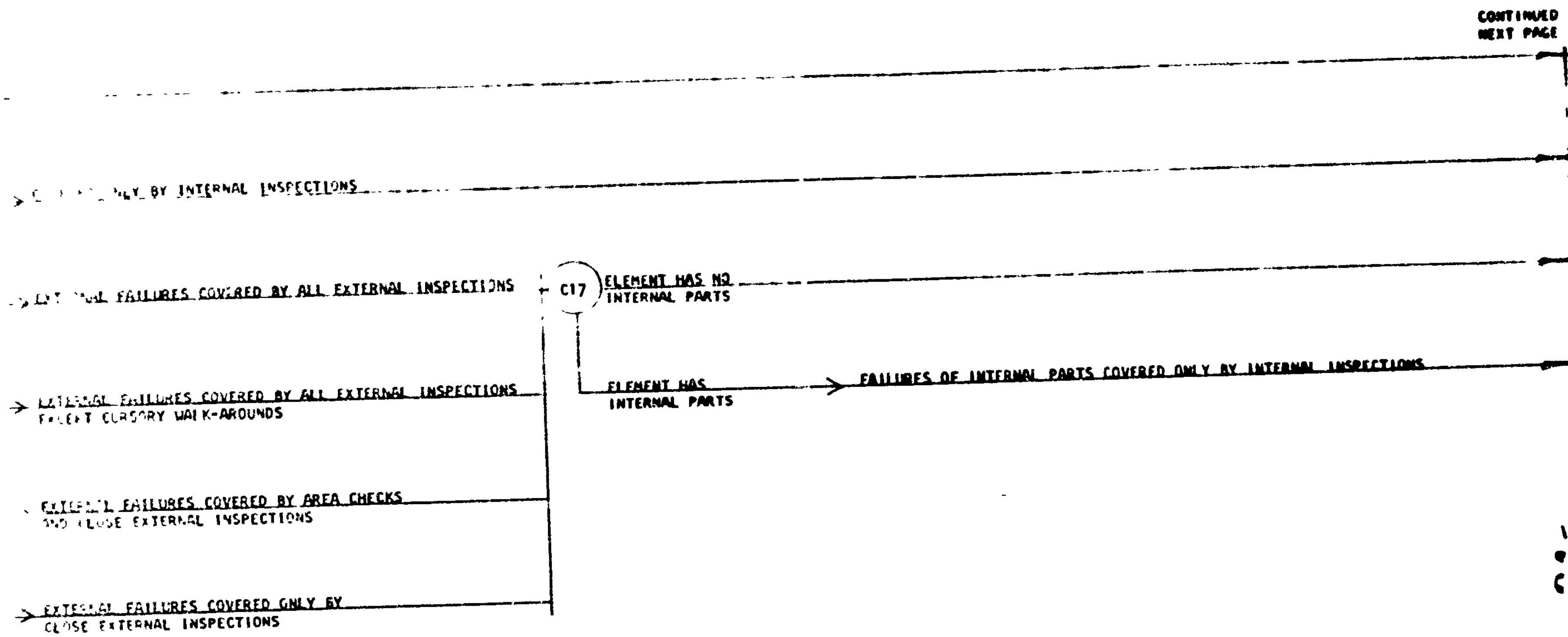
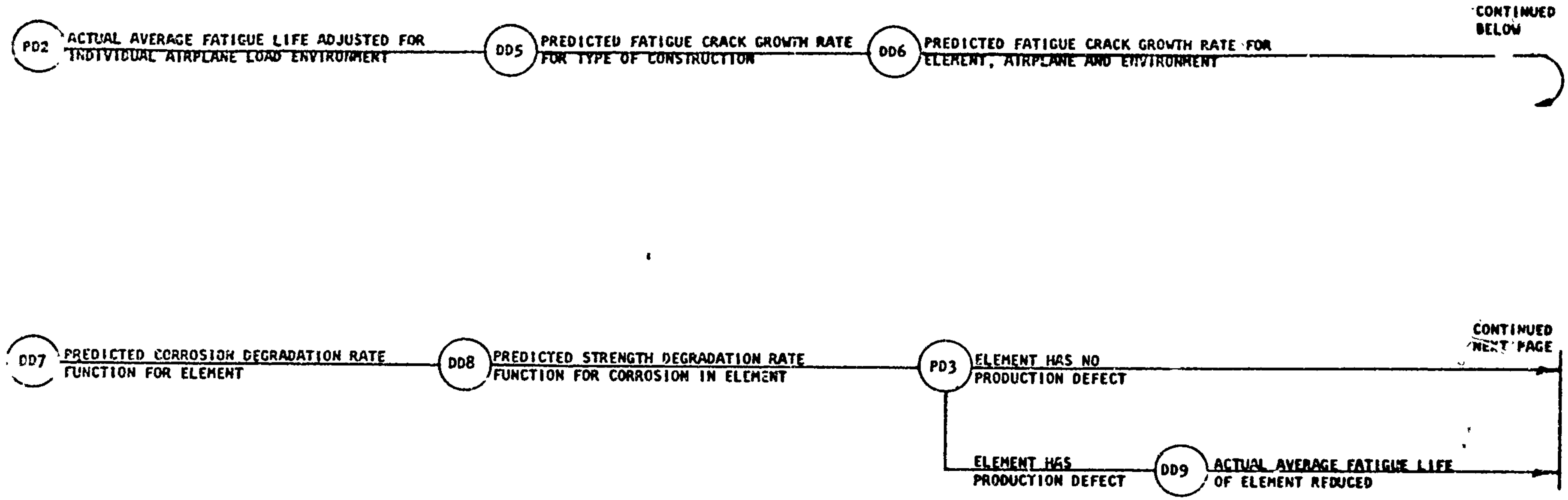
003
INSPECTIONS INCREASED PENDING MODIFICATION
C10
MODIFICATION TESTED
C1
ACTUAL AVERAGE MODIFIED FATIGUE LIFE
MODIFICATION NOT TESTED
C1
PREDICTED AVERAGE MODIFIED FATIGUE LIFE
PD1
ACTUAL AVERAGE MODIFIED FATIGUE LIFE
DS2
MODIFICATION NOT AVAILABLE AND NOT INSTALLED
MODIFICATION AVAILABLE AND INSTALLED
ACTUAL AVERAGE MODIFIED FATIGUE LIFE



OF ELEMENT
CONTINUED NEXT PAGE
DETERMINED
C13
HAVE EXTERNAL PARTS
C14
FAILURE OF EXTERNAL PARTS WOULD CAUSE FUEL LEAK
FAILURE OF EXTERNAL PARTS WOULD NOT CAUSE FUEL LEAK
C15
ELEMENT UNDER 10 FEET FROM GROUND
ELEMENT OVER 10 FEET FROM GROUND
C16
ELEMENT CRITICAL AND EASILY ACCESSIBLE
ELEMENT NOT CRITICAL AND EASILY ACCESSIBLE



CONTINUED NEXT PAGE
BY INTERNAL INSPECTIONS
FAILURES COVERED BY ALL EXTERNAL INSPECTIONS
C17
ELEMENT HAS NO INTERNAL PARTS
ELEMENT HAS INTERNAL PARTS
FAILURES OF INTERNAL PARTS COVERED ONLY BY INTERNAL INSPECTIONS
EXTERNAL FAILURES COVERED BY ALL EXTERNAL INSPECTIONS
CURSORY WALK-AROUNDS
FAILURES COVERED BY AREA CHECKS
EXTERNAL INSPECTIONS
EXTERNAL FAILURES COVERED ONLY BY
CLOSE EXTERNAL INSPECTIONS


PD2
ACTUAL AVERAGE FATIGUE LIFE ADJUSTED FOR INDIVIDUAL AIRPLANE LOAD ENVIRONMENT
DD5
PREDICTED FATIGUE CRACK GROWTH RATE FOR TYPE OF CONSTRUCTION
DD6
PREDICTED FATIGUE CRACK GROWTH RATE FOR ELEMENT, AIRPLANE AND ENVIRONMENT
CONTINUED BELOW
DD7
PREDICTED CORROSION DEGRADATION RATE FUNCTION FOR ELEMENT
DD8
PREDICTED STRENGTH DEGRADATION RATE FUNCTION FOR CORROSION IN ELEMENT
PD3
ELEMENT HAS NO PRODUCTION DEFECT
ELEMENT HAS PRODUCTION DEFECT
DD9
ACTUAL AVERAGE FATIGUE LIFE OF ELEMENT REDUCED
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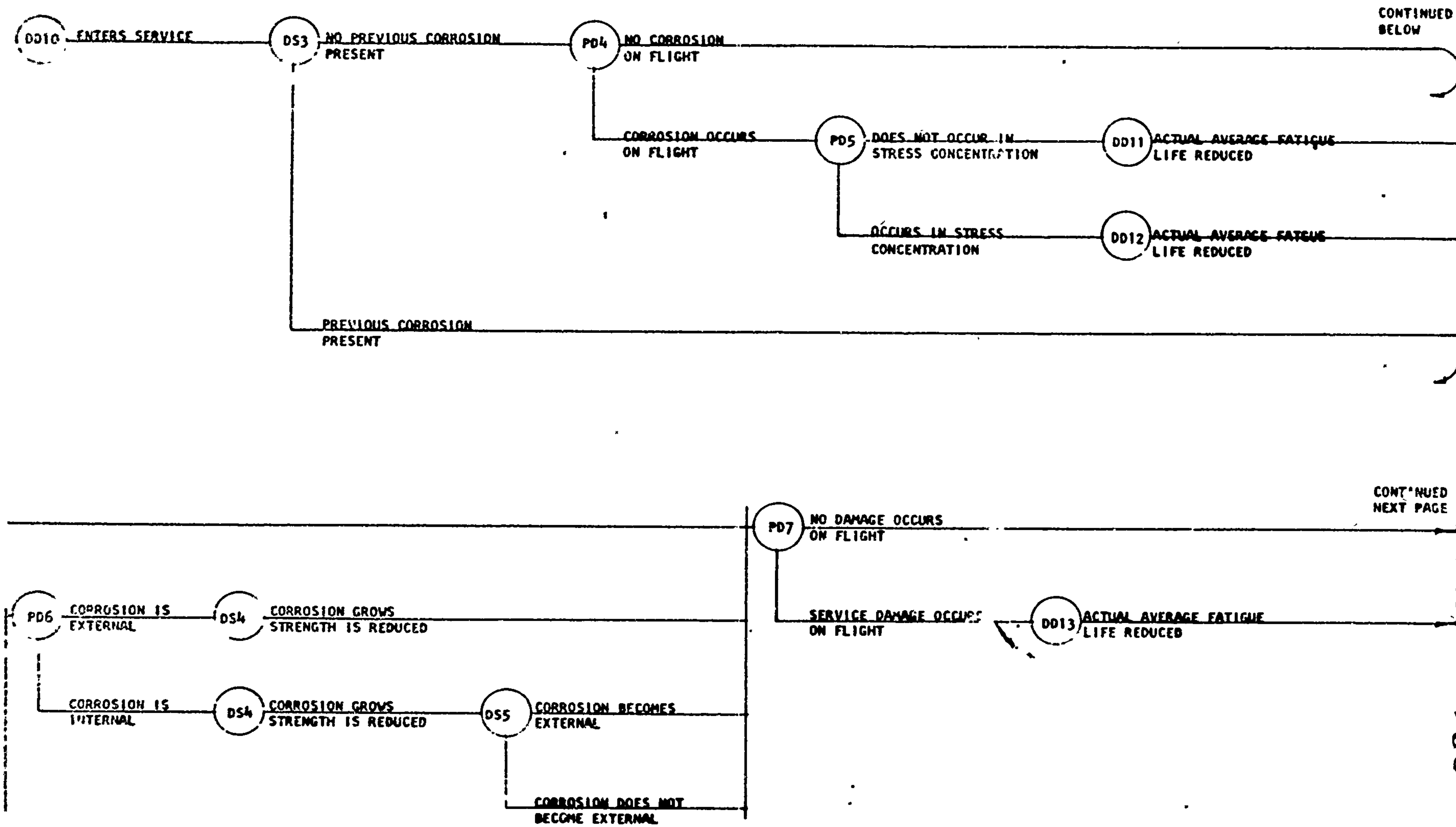
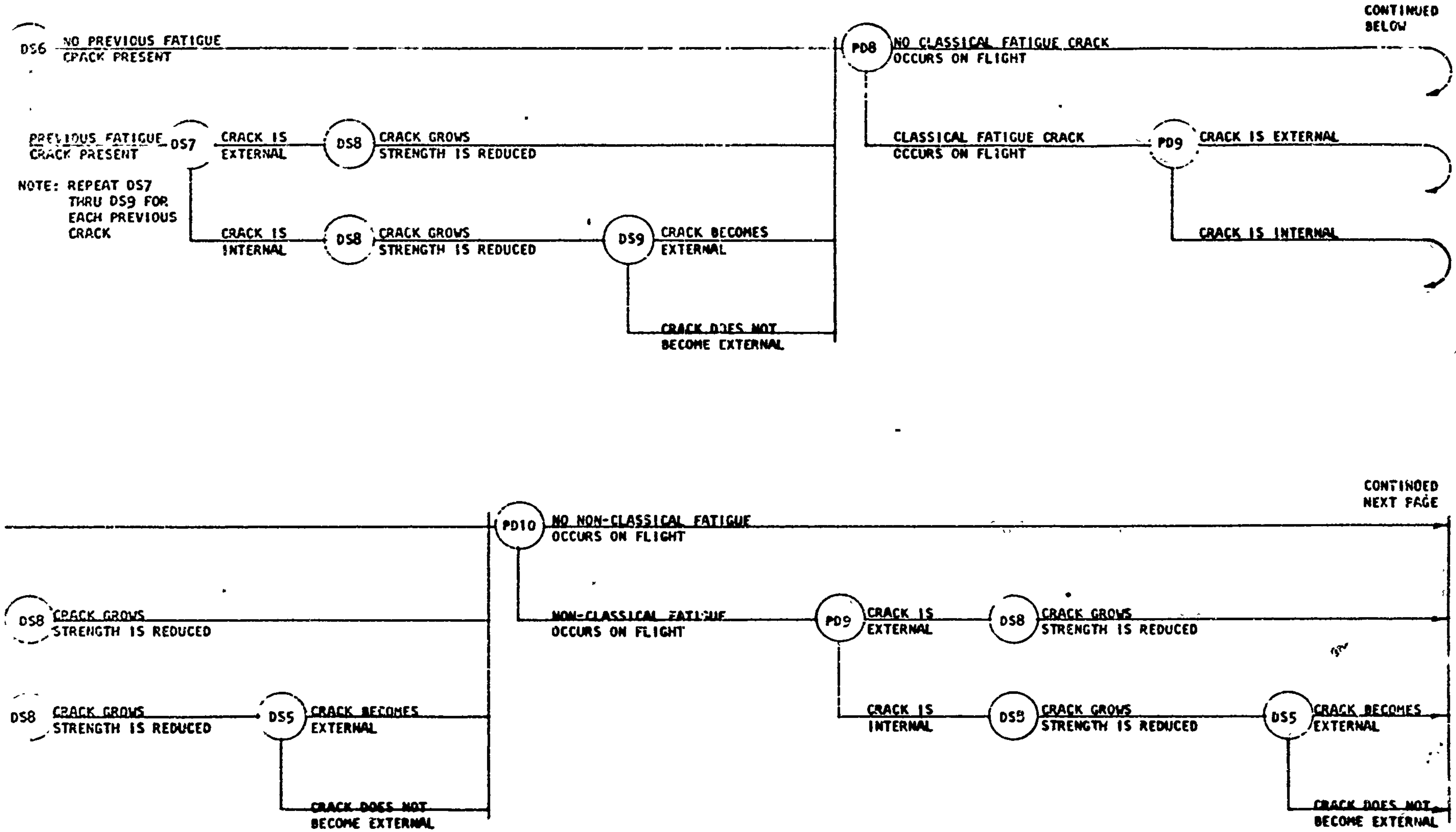

DD10
ENTERS SERVICE
DS3
NO PREVIOUS CORROSION PRESENT
PD4
NO CORROSION ON FLIGHT
CONTINUED BELOW
CORROSION OCCURS ON FLIGHT
PD5
DOES NOT OCCUR IN STRESS CONCENTRATION
DD11
ACTUAL AVERAGE FATIGUE LIFE REDUCED
OCCURS IN STRESS CONCENTRATION
DD12
ACTUAL AVERAGE FATIGUE LIFE REDUCED
PREVIOUS CORROSION PRESENT
PD7
NO DAMAGE OCCURS ON FLIGHT
CONT'NUED NEXT PAGE
SERVICE DAMAGE OCCURS ON FLIGHT
DD13
ACTUAL AVERAGE FATIGUE LIFE REDUCED
PD6
CORROSION IS EXTERNAL
DS4
CORROSION GROWS STRENGTH IS REDUCED
CORROSION IS INTERNAL
DS4
CORROSION GROWS STRENGTH IS REDUCED
DS5
CORROSION BECOMES EXTERNAL
CORROSION DOES NOT BECOME EXTERNAL


DS6
NO PREVIOUS FATIGUE CRACK PRESENT
PREVIOUS FATIGUE CRACK PRESENT
DS7
NOTE: REPEAT DS7 THRU DS9 FOR EACH PREVIOUS CRACK
CRACK IS EXTERNAL
CRACK IS INTERNAL
DS8
CRACK GROWS STRENGTH IS REDUCED
DS8
CRACK GROWS STRENGTH IS REDUCED
DS9
CRACK BECOMES EXTERNAL
CRACK DOES NOT BECOME EXTERNAL
PD8
NO CLASSICAL FATIGUE CRACK OCCURS ON FLIGHT
CLASSICAL FATIGUE CRACK OCCURS ON FLIGHT
PD9
CRACK IS EXTERNAL
CRACK IS INTERNAL
CONTINUED BELOW
DS8
CRACK GROWS STRENGTH IS REDUCED
DS8
CRACK GROWS STRENGTH IS REDUCED
DS5
CRACK BECOMES EXTERNAL
CRACK DOES NOT BECOME EXTERNAL
PD10
NO NON-CLASSICAL FATIGUE OCCURS ON FLIGHT
NON-CLASSICAL FATIGUE OCCURS ON FLIGHT
PD9
CRACK IS EXTERNAL
CRACK IS INTERNAL
DS8
CRACK GROWS STRENGTH IS REDUCED
DS9
CRACK GROWS STRENGTH IS REDUCED
DS5
CRACK BECOMES EXTERNAL
CRACK DOES NOT BECOME EXTERNAL
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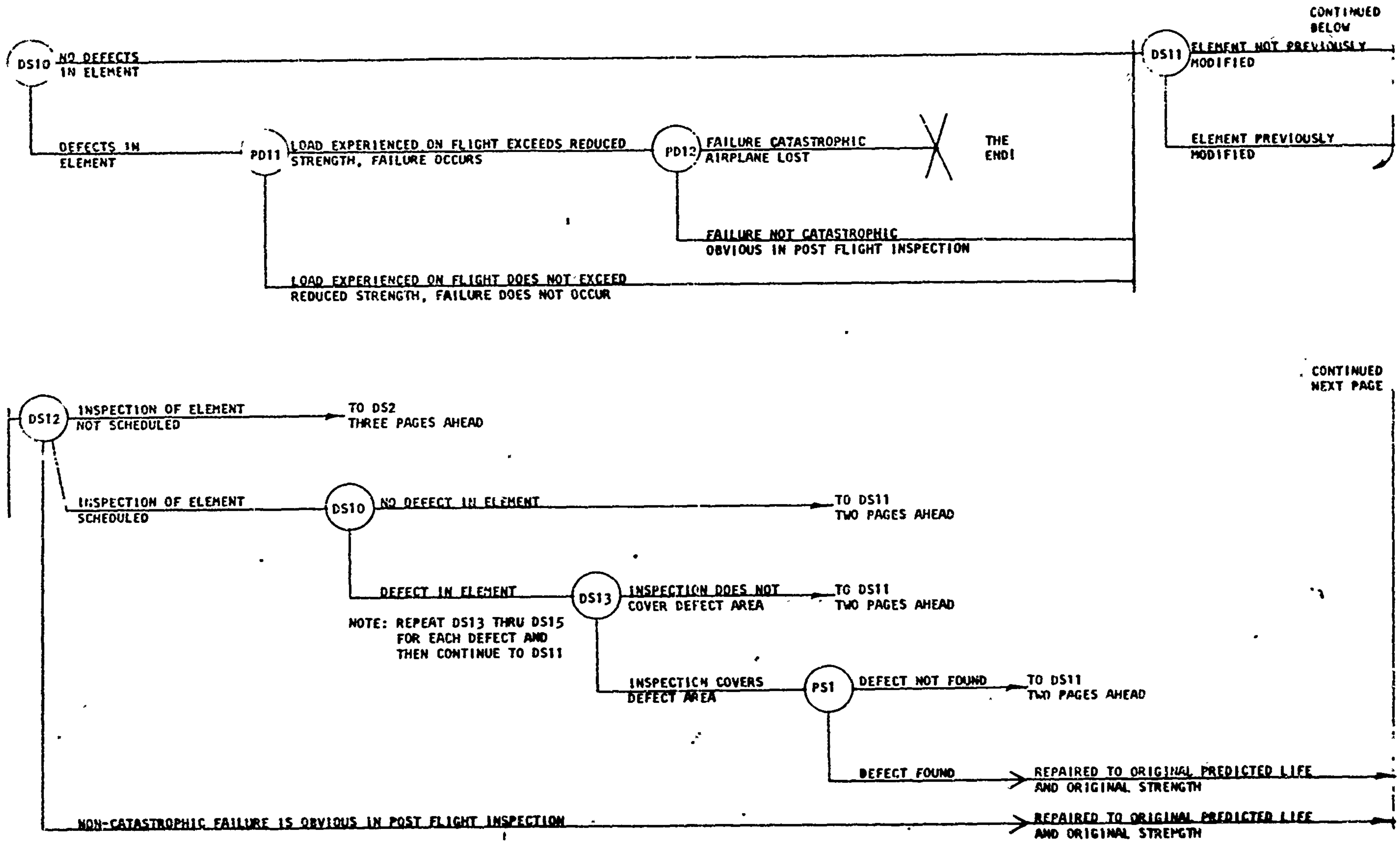
DS10
NO DEFECTS IN ELEMENT
DEFECTS IN ELEMENT
PD11
LOAD EXPERIENCED ON FLIGHT EXCEEDS REDUCED STRENGTH, FAILURE OCCURS
LOAD EXPERIENCED ON FLIGHT DOES NOT EXCEED REDUCED STRENGTH, FAILURE DOES NOT OCCUR
PD12
FAILURE CATASTROPHIC AIRPLANE LOST
THE END!
FAILURE NOT CATASTROPHIC OBVIOUS IN POST FLIGHT INSPECTION
DS11
ELEMENT NOT PREVIOUSLY MODIFIED
ELEMENT PREVIOUSLY MODIFIED
CONTINUED BELOW
DS12
INSPECTION OF ELEMENT NOT SCHEDULED
TO DS2 THREE PAGES AHEAD
INSPECTION OF ELEMENT SCHEDULED
DS10
NO DEFECT IN ELEMENT
TO DS11 TWO PAGES AHEAD
DEFECT IN ELEMENT
NOTE: REPEAT DS13 THRU DS15 FOR EACH DEFECT AND THEN CONTINUE TO DS11
DS13
INSPECTION DOES NOT COVER DEFECT AREA
TO DS11 TWO PAGES AHEAD
INSPECTION COVERS DEFECT AREA
PS1
DEFECT NOT FOUND
TO DS11 TWO PAGES AHEAD
DEFECT FOUND
REPAIRED TO ORIGINAL PREDICTED LIFE AND ORIGINAL STRENGTH
NON-CATASTROPHIC FAILURE IS OBVIOUS IN POST FLIGHT INSPECTION
REPAIRED TO ORIGINAL PREDICTED LIFE AND ORIGINAL STRENGTH
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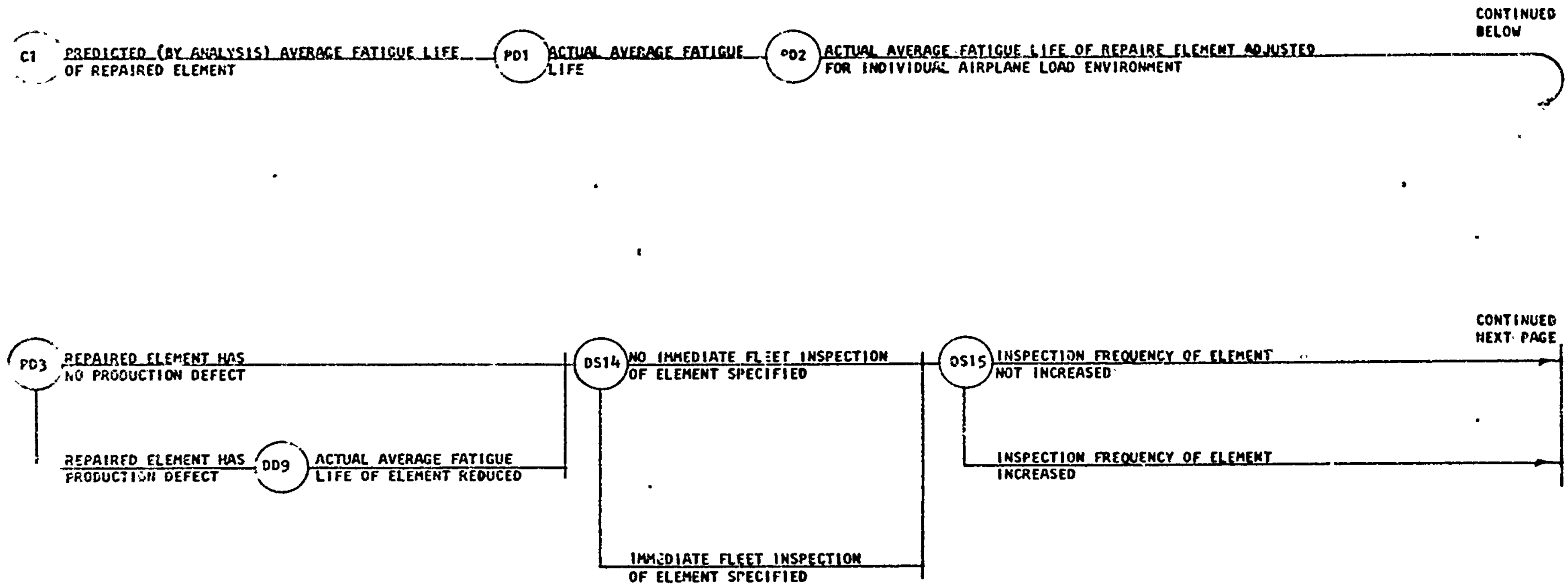
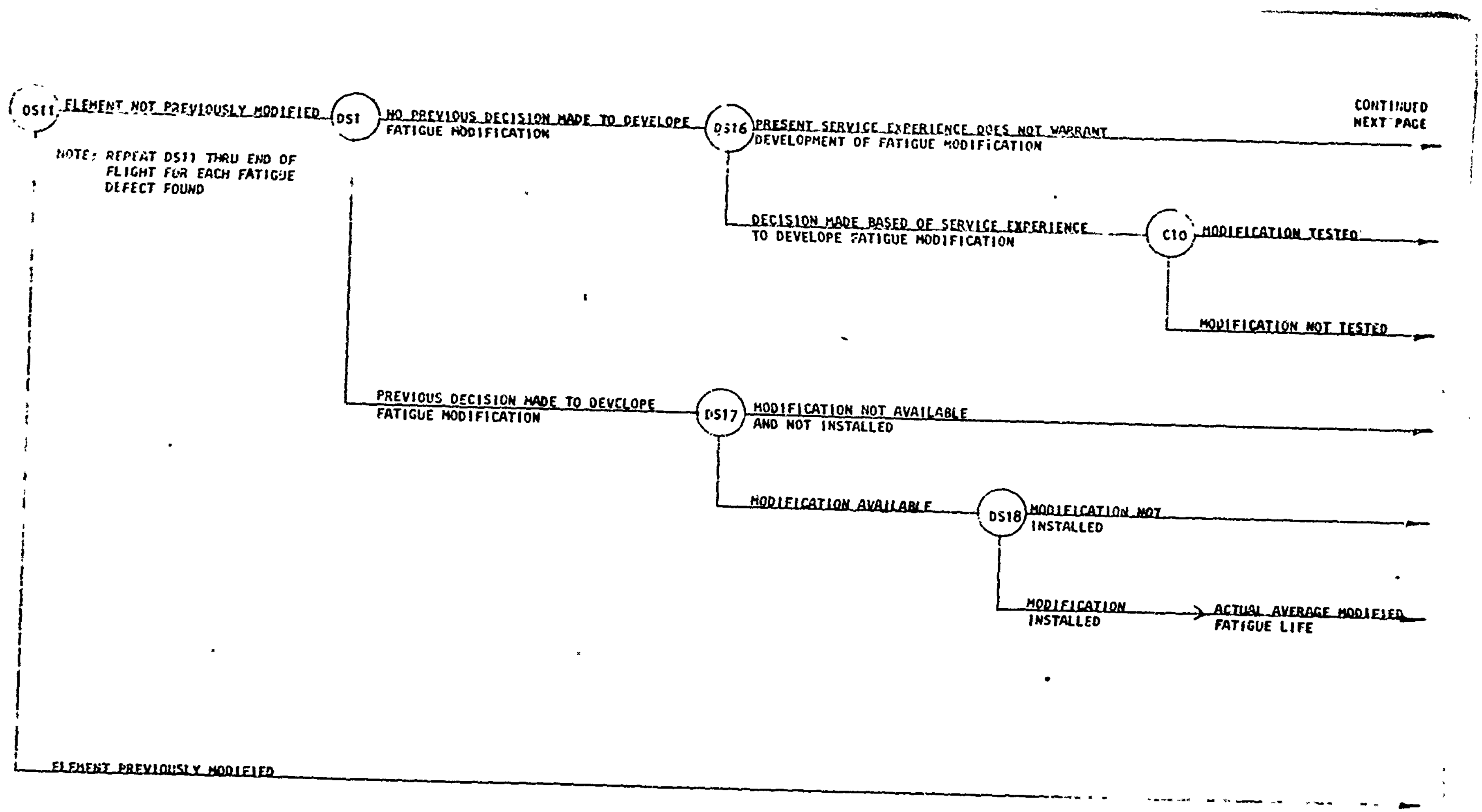
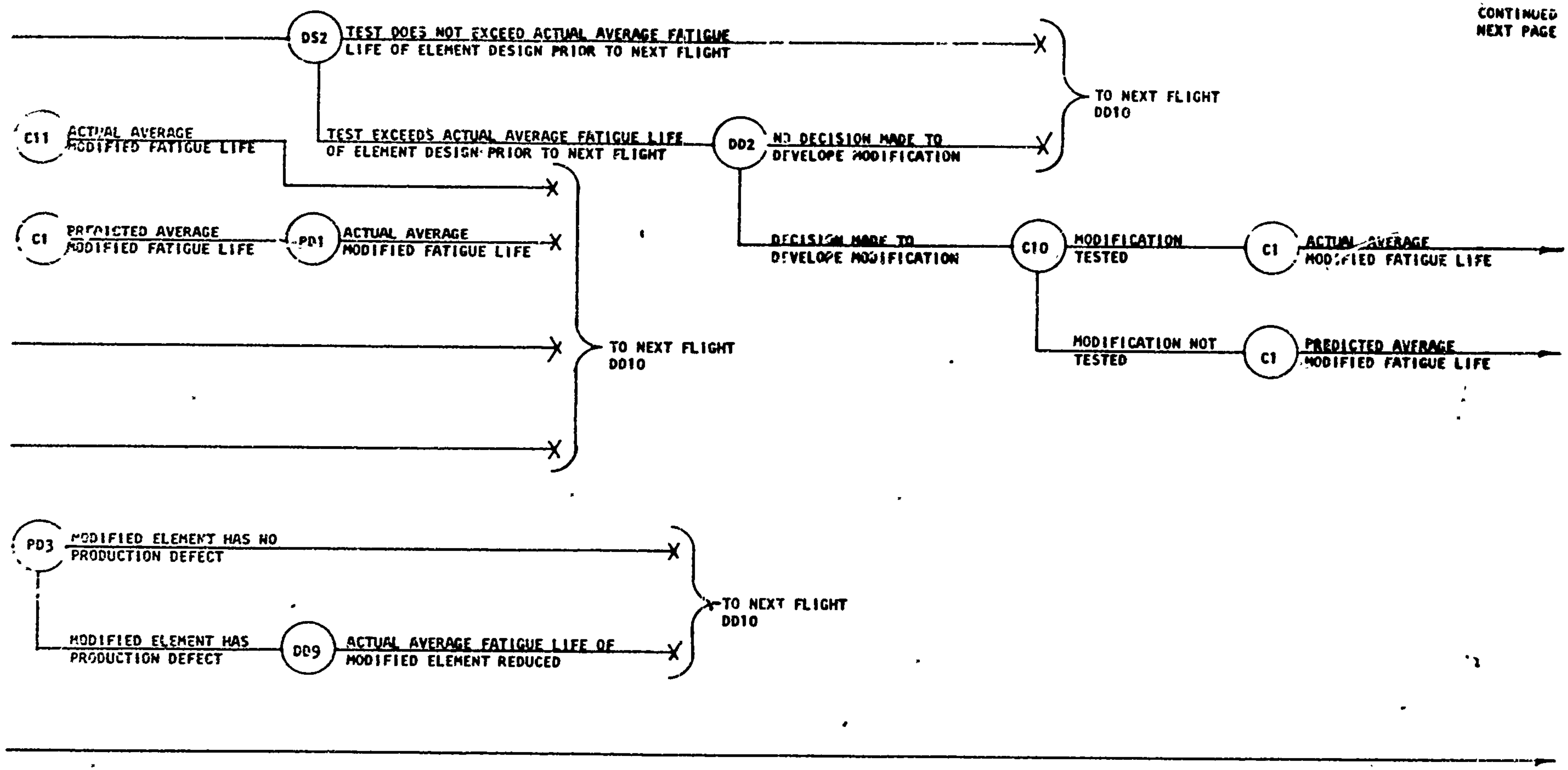


C1
PREDICTED (BY ANALYSIS) AVERAGE FATIGUE LIFE OF REPAIRED ELEMENT
PD1
ACTUAL AVERAGE FATIGUE LIFE
PD2
ACTUAL AVERAGE FATIGUE LIFE OF REPAIRE ELEMENT ADJUSTED FOR INDIVIDUAL AIRPLANE LOAD ENVIRONMENT
CONTINUED BELOW
PD3
REPAIRED ELEMENT HAS NO PRODUCTION DEFECT
REPAIRED ELEMENT HAS PRODUCTION DEFECT
DD9
ACTUAL AVERAGE FATIGUE LIFE OF ELEMENT REDUCED
DS14
NO IMMEDIATE FLEET INSPECTION OF ELEMENT SPECIFIED
IMMEDIATE FLEET INSPECTION OF ELEMENT SPECIFIED
DS15
INSPECTION FREQUENCY OF ELEMENT NOT INCREASED
INSPECTION FREQUENCY OF ELEMENT INCREASED
CONTINUED NEXT PAGE


DS11
ELEMENT NOT PREVIOUSLY MODIFIED
DS1
NO PREVIOUS DECISION MADE TO DEVELOPE FATIGUE MODIFICATION
DS16
PRESENT SERVICE EXPERIENCE DOES NOT WARRANT DEVELOPMENT OF FATIGUE MODIFICATION
CONTINUED NEXT PAGE
DECISION MADE BASED OF SERVICE EXPERIENCE TO DEVELOPE FATIGUE MODIFICATION
C10
MODIFICATION TESTED
MODIFICATION NOT TESTED
NOTE: REPEAT DS11 THRU END OF FLIGHT FOR EACH FATIGUE DEFECT FOUND
PREVIOUS DECISION MADE TO DEVELOPE FATIGUE MODIFICATION
DS17
MODIFICATION NOT AVAILABLE AND NOT INSTALLED
MODIFICATION AVAILABLE
DS18
MODIFICATION NOT INSTALLED
MODIFICATION INSTALLED
ACTUAL AVERAGE MODIFIED FATIGUE LIFE
ELEMENT PREVIOUSLY MODIFIED



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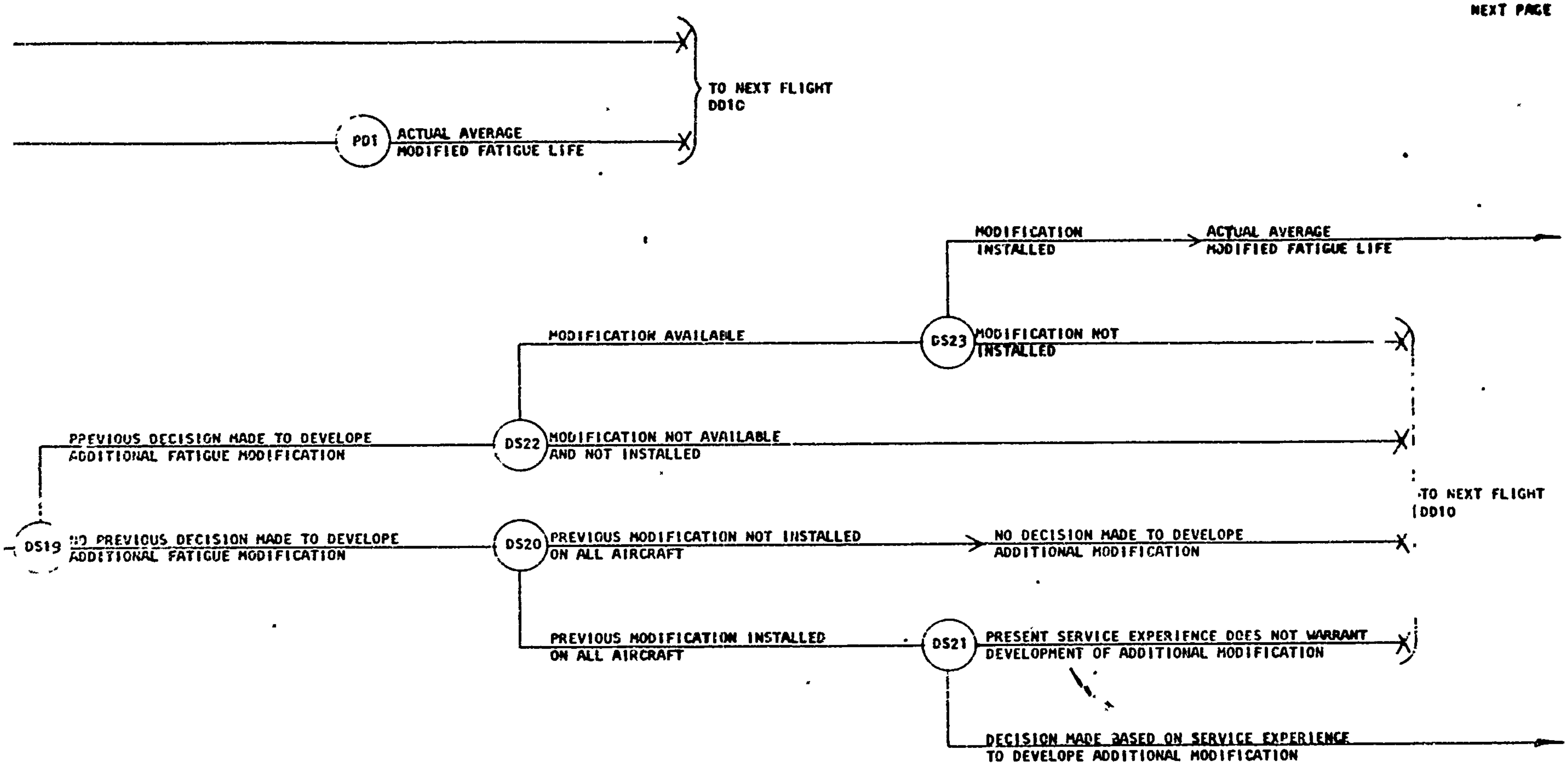

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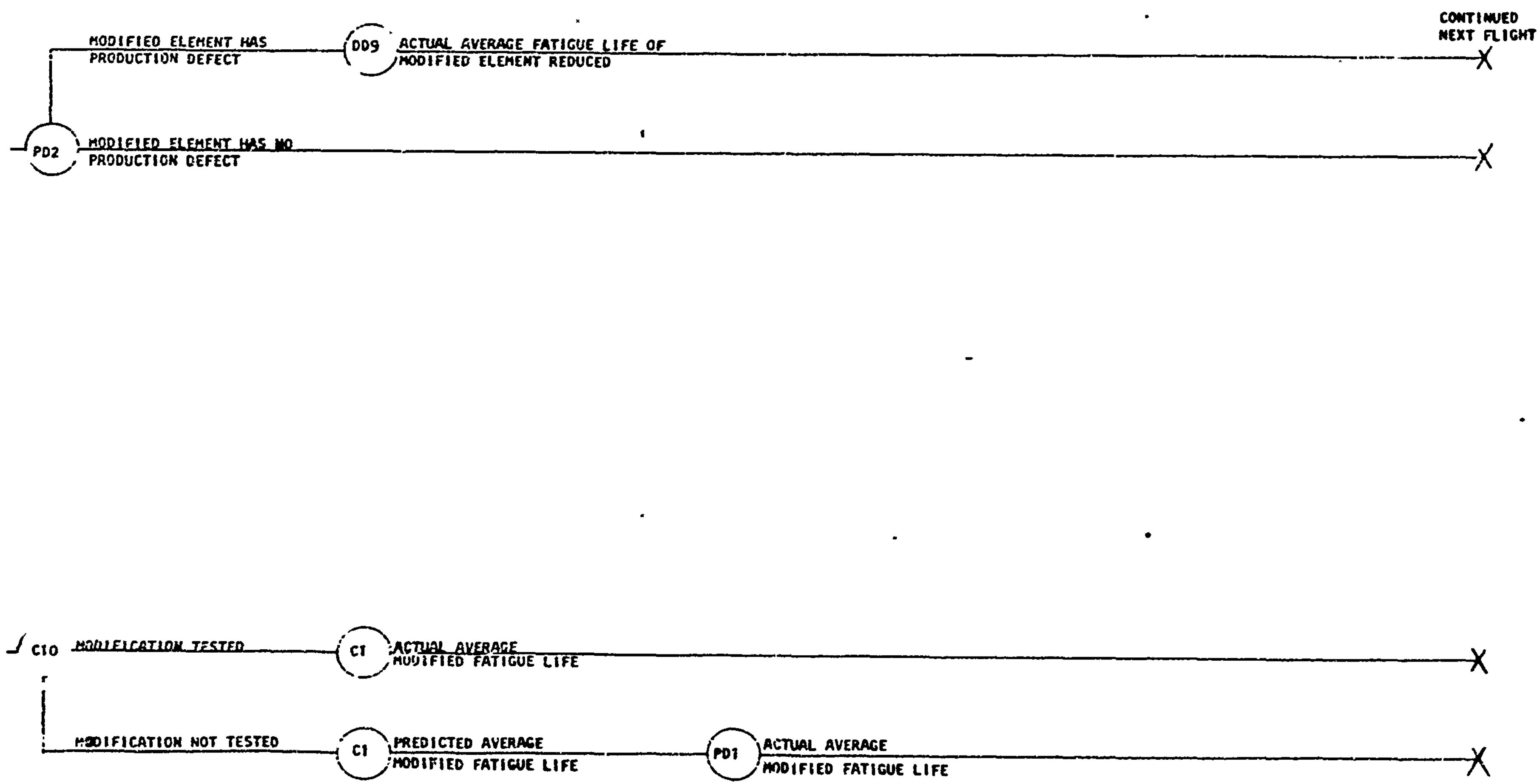
MODIFIED ELEMENT HAS PRODUCTION DEFECT
DD9
ACTUAL AVERAGE FATIGUE LIFE OF MODIFIED ELEMENT REDUCED
CONTINUED NEXT FLIGHT
PD2
MODIFIED ELEMENT HAS NO PRODUCTION DEFECT
C10
MODIFICATION TESTED
C1
ACTUAL AVERAGE MODIFIED FATIGUE LIFE
MODIFICATION NOT TESTED
C1
PREDICTED AVERAGE MODIFIED FATIGUE LIFE
PD1
ACTUAL AVERAGE MODIFIED FATIGUE LIFE

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